NASA CR-66608

LUNAR ORBITER V

GP PRICE $ ______

CF PRICE(S) $ ______

Hard copy (HC) 3.00

Microfiche (MF) .65

ff July 65

MISSION SYSTEM PERFORMANCE

N 68 - 19923 (ACCESSION NUMBER)

143 (PAGES)

CR 66608 (NASA CR OR TMX OR AD NUMBER)

1 (THRU)

(CODE)

31 (CATEGORY)

FACILITY FORM 602

Prepared for the
NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

Langley Research Center

by

THE BOEING COMPANY

Contract No. NAS1—3800

January 4, 1968

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BOEING DOCUMENT D2-100755-3

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Foreword

The Lunar Orbiter V final report is divided into six volumes as follows.

Volume I	Mission Summary
Volume II	Photography
Volume III	Mission System Performance
Volume IV	Extended-Mission Spacecraft and Subsystem Performance
Volume V	Deleted
Volume VI	Appendices
Volume VII	Postmission Photo Supporting Data

Volume I summarizes the photographic mission concepts and conduct, system performance, and results. Volume II contains the mission photographic planning and conduct and a description and analysis of the photos obtained during Mission V together with photo supporting data. Volume III contains a discussion and performance analysis of the Lunar Orbiter and its subsystems. It also includes launch operations, flight conduct and flight path control information, as well as discussions of the anomalies encountered during the mission. Volume IV summarizes the operational reports covering the extended mission and contains a discussion and analysis of spacecraft performance, experiments conducted, and anomalies encountered during the extended mission. Volume V has been deleted and the information similar to that contained in Volume V of earlier missions has been included in Volume IV. Volume VI contains selected detail data and information in support of the analyses presented in Volumes II and III. Volume VII provides data on computer programs and requirements to compute photo positioning and orientation data. It also includes a complete EVAL computer printout of each photo readout.

Lunar Orbiter hardware descriptions will be found in the Lunar Orbiter I Final Report.

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1.0 Launch Operations

The Launch Operations Plan for the Lunar Orbiter Project (LOP), Lockheed Missiles and Space Company Document LMSC - A751901D, as modified by LMSC letter "Launch Information Letter for Lunar Orbiter, Mission 5 (Agena Vehicle 6634)," LMSC-A881381/66-51/537 dated July 21, 1967, and LMSC "Final Launch Criteria TWX, Agena Vehicle 6634, Lunar Orbiter Mission 5," LMSC-A882359, dated July 24, 1967, provided the primary planning for overall space vehicle program direction through the lunar preinjection phase of the Lunar Orbiter V flight. This document served as the basis for directing the activities required to achieve and evaluate flight objectives, launch criteria, and constraints; and implement space vehicle preflight tests, checkouts, and launch.

The same basic launch operations plan used for the Mission V launch was used during the first four missions. A description of the launch operation organization and supporting launch/postlaunch tracking and communication facilities is contained in the NASA Document CR 66324 *Lunar Orbiter Mission I Final Report*, Volume III Section 3.3.1, "Launch Operation Plan" and 3.3.2, "Launch Base Facilities."

1.1 SPACECRAFT PROCESSING

Spacecraft 3 arrived at Cape Kennedy on March 10, 1967, to be prepared for use as a backup for the Mission IV flight article, Spacecraft 7. Upon arrival, it was moved to Hangar "S" to initiate processing for the backup function. This spacecraft was accepted by NASA on April 23. On April 25 Spacecraft 3 was taken to the Explosive Safe Area to be fueled and fully prepared for flight. At this time a leaking fill and test valve was discovered and a new valve was installed on the spacecraft. Encapsulation of this spacecraft provided the capability, if necessary, of exchanging spacecraft and supporting a launch date of May 5, 1967.

Following the launch of Spacecraft 7 on May 4, 1967, deactivation of Spacecraft 3 was started in preparation for storage until needed for Mission V. However, during defueling a leak in the oxidizer tank bladders was detected that was more than the 32 cubic centimeters per hour allowable. The spacecraft was then shipped to Seattle by airconditioned van for removal and replacement of the oxidizer tanks. A complete leak check was run on the tubing and bladders at this time.

1.1.1 Hangar "S"

On June 23, 1967, Spacecraft 3 arrived back at Cape Kennedy for preflight testing per the original complete Hangar "S" test to which all spacecraft were tested following their arrival at the Cape (see Table I-1). The retest document that was used for testing previous spacecraft after storage following their backup function was not used for retesting Spacecraft 3. These complete retests were performed to ascertain that all subsystems were still satisfactory after cross-country travel and to test those subsystems that were modified for Mission V. All retests were satisfactorily concluded and no significant discrepancies were found. Refer to Table 1-2 for a summary of differences between Lunar Orbiter IV and V. However, two micrometeoroid detectors, Serial Numbers 237 and 315, were replaced when visual examination revealed deep scratches on them. Also, the inertial reference unit (IRU), Serial Number 108, which had been shipped to Seattle for component level tests, was replaced with IRU Serial Number 110 when an air bubble was detected in the accelerometer.

1.1.2 Explosive Safe Area

On July 13, 1967, the spacecraft was moved to the Explosive Safe Area (ESA) for flight fueling and final testing. A listing of tests performed at the ESA is shown in Table 1-3. After the regulator and leak check, fueling was started on June 15, 1967.

After the DSIF-71 test without shroud on June 20, the Agena adapter, thermal barrier, and shroud were installed. On June 25, following the DSIF-71 test with shroud, the encapsulated spacecraft was moved to Pad 13.

Table 1-1: Hangar S Retests
Spacecraft Alignment Verification
Pre-"Power On" Check
Initial Test Setup
Initial Systems Status Verification
Communications Performance Test
Radiation Dosage Scintillation Counter FCO
Attitude Control Functional Test
Velocity Control Subsystem
Power Subsystem Performance
High-Gain-Antenna Position Control, Camera Thermal Door Operation, and Antenna Deployment
Solar Panel Test and Low-Gain-Antenna Alignment
Sun Sensor Alignment Verification
Photo Subsystem Functional Checkout
Photo Subsystem Removal
Equipment Mounting Deck Reflectance Test
Camera 610-mm-Shutter Test
TWTA Power Output Telemetry Calibration
Star Tracker Mapping Voltage Calibration
Transponder AGC Calibration
Mode I Modulation Index (498-kc Index)

Note: "Spacecraft/Hangar S/DSIF-71 Checkout Test" was also run as a function of Hangar S retests.

Table 1-2: Summary of Differences Between Lunar Orbiters IV and V

Photo Subsystem
- Provided electrical rather than mechanical adjustment of 610-mm shutter slit servo damper.
- Shroud between V/H sensor and back of mirror was changed from aluminum to cloth to decrease mirror vibration.
- Modified the light baffles on the 80- and 610-mm lens and the upper shell.
- Deleted readout-looper-full logic.

Structure and Mechanisms Subsystem
- Increased the number of solar reflectors under the TWTA and photo subsystem.
- Revised the thermal test paint coupons to provide data on Z-93, Hughes' Organic White, S-13G/B-1056, and clear silicone over polished aluminum. See description of this test in Section 3.6.4.2.
- Installed a closed-door position indicator on the camera thermal door with associated telemetry.

Communications Subsystem
- Each spacecraft has a "one of a kind" command decoder address plug.

Power Subsystem
- Added a d.c. voltage boost regulator to provide the proper voltage to the photo subsystem during spacecraft off-Sun operation.
- Modified charge controller to raise the maximum charge rate to 1.4 amps from 1.0 amp.
- Installed a sun shade on the back of the resistor panel and painted areas of Solar Panels 2 and 4 that were shown to contribute glint to the Canopus star tracker.

Attitude Control Subsystem
- Mission IV IRU incorporated Kearfott gyros while Mission V IRU had Sperry gyros.

Table 1-3:
Explosive Safe Area Tests

Photo Subsystem Launch Preparation
Spacecraft Regulator and Leak Check
Propellant Servicing
Nitrogen Servicing
Photo Subsystem Installation and Alignment
Weight and Balance Verification
Battery Verification
Camera Thermal Door Verification
Spacecraft Operational Check (without shroud installed) with DSIF-71
Ordnance Check and Hookup
Agena Adapter Installation
Thermal Barrier Installation
Nose Fairing Installation
*Spacecraft Operational Check (with shroud installed) with DSIF-71
*Transport Spacecraft (ESA to Pad)

Note: All paragraphs except those marked * were also previously run during Mission IV backup tests.

1.1.3 Launch Pad 13

Matchmate of the Lunar Orbiter spacecraft to the Atlas-Agena launch vehicle with associated impedance and interface tests took place without incident. Spacecraft tests conducted on the launch pad are shown in Table 1-4. Upon completion of the initial pad tests, the spacecraft was ready for simulated launch.

1.2 LAUNCH CONDUCT

The launch plan, activities, facilities, and participating organizations were similar to those for the previous missions. Specific information may be obtained from Section 3.3, "Launch Operations," of the *Lunar Orbiter I Final Report*.

1.2.1 Launch Criteria

Launch criteria and space vehicle preparation were governed by the *Launch Operations Plan for the Lunar Orbiter Project*, LMSC/A 751901D. Although Spacecraft 3 had been tested and used as a backup to Spacecraft 7 for Mission IV, it was necessary to completely retest it for Mission V due to the leaking oxidizer tanks, which had been replaced in Seattle.

Significant milestones described in Table 1-5 were satisfactorily completed by Spacecraft 3 in preparation for launch.

1.2.2 Countdowns and Launch

The spacecraft did not participate in the joint flight acceptance composite test (J-FACT) on July 24, 1967, for Mission V. During the plus count of the test, an Agena range safety command battery failed. Further testing confirmed that the battery was faulty. Several voltage level changes were observed on the Atlas 400-cycle three-phase output at about T+100. This phenomenon has been observed previously and is attributed to interaction between the gyro heater mag-amps and the inverter. One Agena monitor indication was not received and was found to be caused by a broken wire in a ground equipment connector.

The simulated launch test was conducted July 28, 1967. All test objectives were met. The following problems were encountered.

- The landline transducer for measurement F1304P, staging bottle pressure, was replaced when an intermittent reading was observed.
- A low hydraulic pressure indication was noted on the remote pressure gauge for the Agena boom. Investigation disclosed

a bad connector and transducer in the system. Both items were replaced and the system was satisfactorily checked out.

- At T-430 minutes spacecraft power turn-on was delayed approximately 45 minutes due to switch panel configuration at Tel-4.

Table 1-4: Launch Area Test Summary

Spacecraft, Adapter, and Agena Matchmate
Lunar Orbiter Spacecraft - Fifth Flight Spacecraft - Initial Pad Tests
Lunar Orbiter Spacecraft - Fifth Flight Spacecraft - Simulated Launch
Lunar Orbiter Spacecraft - Fifth Flight Spacecraft - Launch Countdown

Table 1-5: Spacecraft Prelaunch Milestones

Date	Event
June 23, 1967	Spacecraft arrival (second time)
July 12, 1967	Spacecraft completion of checks in Hangar S prior to moving to the ESA
July 13, 1967	Spacecraft moved to the ESA
July 15, 1967	Spacecraft fueling
July 20, 1967	DSIF checks without the shroud
July 21, 1967	Spacecraft-to-adapter mate
July 22, 1967	Thermal barrier installed
July 23, 1967	Spacecraft encapsulation
July 24, 1967	Spacecraft checkout with DSIF (shroud on)
July 25, 1967	Spacecraft-to-Agena mate
July 28, 1967	Simulated launch
August 1, 1967	Launch

- The multipath problem, previously encountered during checkout of the spacecraft while setting up the photo system for optimum video, was once again experienced at T-315 minutes. The traveling-wave-tube amplifier (TWTA) on high power, using the repeater antenna, did not produce satisfactory results. The command to turn off the TWTA was not received by the spacecraft due to command transmission problems caused by a low uplink signal. However, a successful method for verification of the photo system operation was devised using high power on the TWTA and readout from the spacecraft van. The delays caused in finding and correcting the signal transmission problem impacted the simulated rf silence period, but no hold was called and the spacecraft was permitted to operate during the simulated rf silent period.
- At T-60 minutes the same problem was experienced with Tel-4 concerning data retransmission. The spacecraft was delayed approximately 7 minutes before beginning internal power checks. From this point the spacecraft encountered no further problems and followed the clock to T-0.

After successful completion of the simulated launch, however, both Atlas airborne command destruct receivers were changed at the request of Range Safety.

The planned launch countdown for August 1, 1967, which was the first day of a 3-day launch period, included two built-in holds, one of 50 minutes duration at T-60 and a second of 10 minutes duration at T-7. However, only by launching on the first day of the period could the desired photography of the Moon's farside be achieved.

The spacecraft count picked up on schedule at T-530 minutes and proceeded without incident until photo readout at T-320 minutes, when the rf multipath problem was again encountered. The flight configuration used allowed optimum video for the photo readout. The spacecraft TWTA was turned off at T-255 minutes just prior to the rf silent period. Spacecraft testing caught up with the schedule at T-215 minutes.

Meanwhile, at T-261, determination was made that the Agena velocity meter required replacement. The count continued, however, down to T-155, at which time a 55-minute hold was called for completion of testing associated with replacement of the velocity meter.

At 17:20 GMT the Pad Safety Office cleared all personnel from the pad because of lightning within 5 miles. Preparations for fueling were completed prior to clearing the pad, which allowed the count to be resumed at 17:29 GMT (T-155).

At T-119 minutes, the MOD III ground station lost air conditioning, causing temperatures to exceed the specified operating levels. The station was returned to operation at T-94 minutes with portable air conditioning units.

It was necessary to hold again at T-90 (18:34 GMT) because operations were behind schedule. This hold was initially estimated at 20 minutes; however, severe weather conditions (heavy rain, lightning, and wind gusts) forced an extension of the hold and required the service tower to be returned and positioned around the vehicle for protection. The count was not resumed until 20:58 GMT giving this hold a duration of 144 minutes.

After resuming count at T-90, all operations progressed normally down to the built-in hold at T-7 minutes, which began at 22:21 GMT. This hold, planned for a duration of 10 minutes, had been shortened to 5 minutes because of the extensive hold time previously used.

The count was resumed at T-7 minutes (22:26 GMT). All operations then progressed normally down to liftoff, which occurred at 22:33:00.338 GMT, with a flight azimuth of 104.8 degrees.

1.2.3 Weather

Upper wind shears were within acceptable limits. At liftoff, the following weather parameters were recorded.

Temperature	75°F
Relative humidity	94 %
Visibility	10 miles, with light rain
Dew point	73°F
Surface winds	4 knots at 180 degrees
Clouds	Overcast with 10/10 cloud cover
Sea-level atmospheric pressure	30.010 inches of mercury

1.2.4 Tracking Coverage

The TDS (tracking and data system) for the near-Earth phase of Lunar Orbiter Mission V included selected resources of the Air Force Eastern Test Range (AFETR), the Manned Space Flight Network (MSFN), the Deep Space Network (DSN), and the NASA Communications System (NASCOM).

Tracking during the launch phase consisted of C-band tracking of the launch vehicle and reception of VHF and S-band telemetry from the launch vehicle and spacecraft, respectively. Figure 1-1 shows AFETR and MSFN uprange coverage and ground track for Mission V. Tracking data provided to AFETR during the launch phase established (1) the Agena parking orbit in real time, (2) the Agena transfer orbit in real time, and (3) launch vehicle performance evaluation. Figure 1-2 shows times of all electronic tracking during Mission V.

1.2.5 Telemetry Coverage

Real-time data were transmitted to the real-time computer system almost continuously from T-7 until spacecraft separation. Figure 1-3 shows the extent of telemetry coverage. Tel-4 data were transmitted to DSS-71 beginning at T-7 minutes. The source was switched to Grand Bahama at T+3 minutes. Data quality appeared to be excellent. At T+7 minutes, the data source, which also was of excellent quality, was switched to Antigua. The switch to Rose Knot data was made at T+13 minutes, 25 seconds. Data from this source were only fair, with several losses of decommutator lock until LOS at approximately T+18 minutes. Ascension data were selected next and were of good quality. The Coastal Crusader data were selected at T+26 minutes and 25 seconds, with good quality data again observed. Pretoria data, which also appeared to be of good quality, were selected at T+31 minutes and continued until spacecraft separation.

Ground track showing locations of the near-Earth network of tracking and telemetry stations is shown in Figure 1-4. Real-time data received from Tananarive showed spacecraft separation. Carnarvon received velocity meter data from the Agena showing the retromaneuver and played back the data in real time. In addition, data from the range ship Sword Knot were available if necessary.

1.2.6 Optics

This launch was supported by 10 metric cameras, 28 engineering sequential cameras, and 26 documentation cameras. Two engineering sequential cameras and four of the documentation cameras did not operate because of the prevailing weather.

1.3 LAUNCH VEHICLE PERFORMANCE

The launch vehicle first stage consisted of an SLV-3 (Atlas), Serial Number 5805. All SLV-3 flight objectives were satisfied. A proper ascent trajectory was attained and all SLV-3 systems performed satisfactorily. Atlas-Agena separation was properly accomplished and good telemetry data were obtained for Atlas systems analysis.

The launch vehicle second stage was an Agena-D, Serial Number 6634. Agena performance was satisfactory throughout the flight. However, Shroud Separation Monitor (Measurement A52) indicated a "pin 2 separated" condition (3.9 v.d.c) at T + 104.0 seconds and stepped to a "shroud separated" condition (4.9 v.d.c) at T+128.8 seconds. Actual shroud separation occurred at T+310.3 seconds, as indicated by normal shroud-separation disturbances on the accelerometers and, as well, by a reported increase in spacecraft signal strength at the time shroud separation was to occur. The indications on Measurement A52 were apparently related to periods of severe vibration, since the first step occurred shortly after the period of maximum g forces and the second step was coincident with booster engine cutoff. A velocity meter cutoff terminated Agena first and second burns. First burn was of 153.125 seconds duration and second burn was 87.49 seconds duration. Agena telemetry yielded good response for analysis of the Agena system.

Significant ascent trajectory events and times in seconds relative to initial vehicle 2-inch motion are covered in Table 1-6.

The configuration of the Atlas-Agena launch vehicle for Mission V was identical to the Mis-

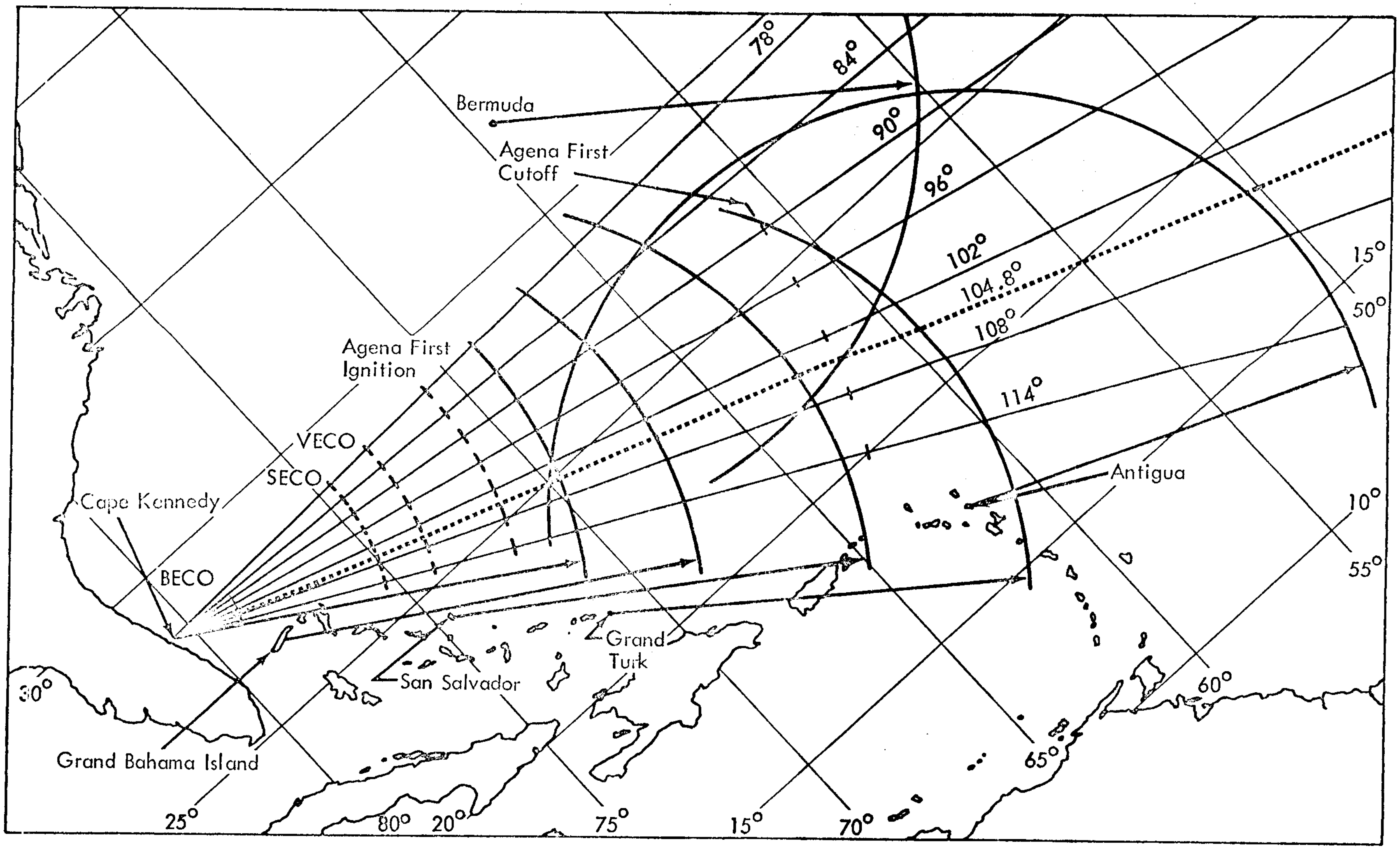


Figure 1-1: Lunar Orbiter Uprange Radar Coverage

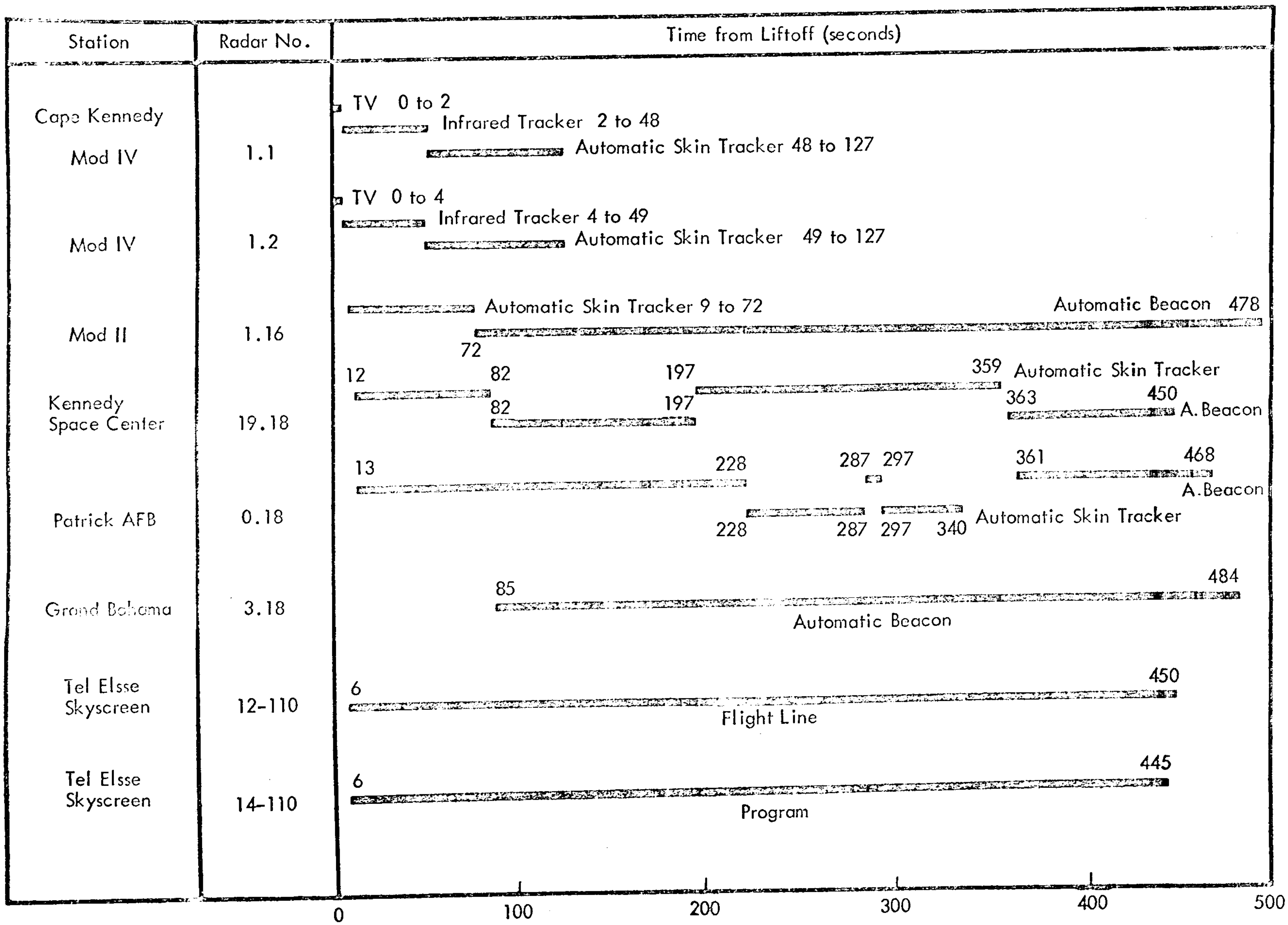


Figure 1-2: Tracking Coverage

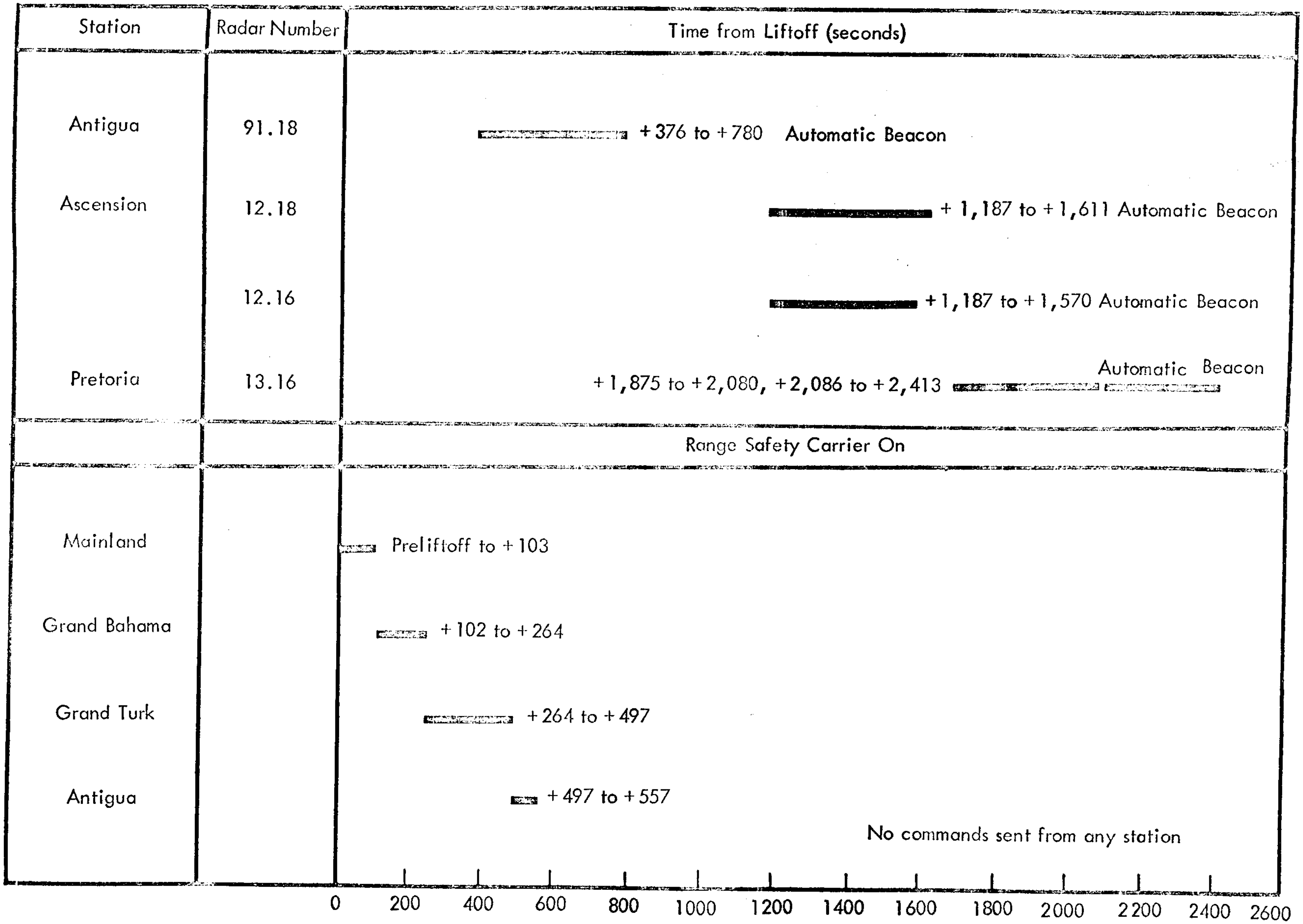


Figure 1-2: (Continued)

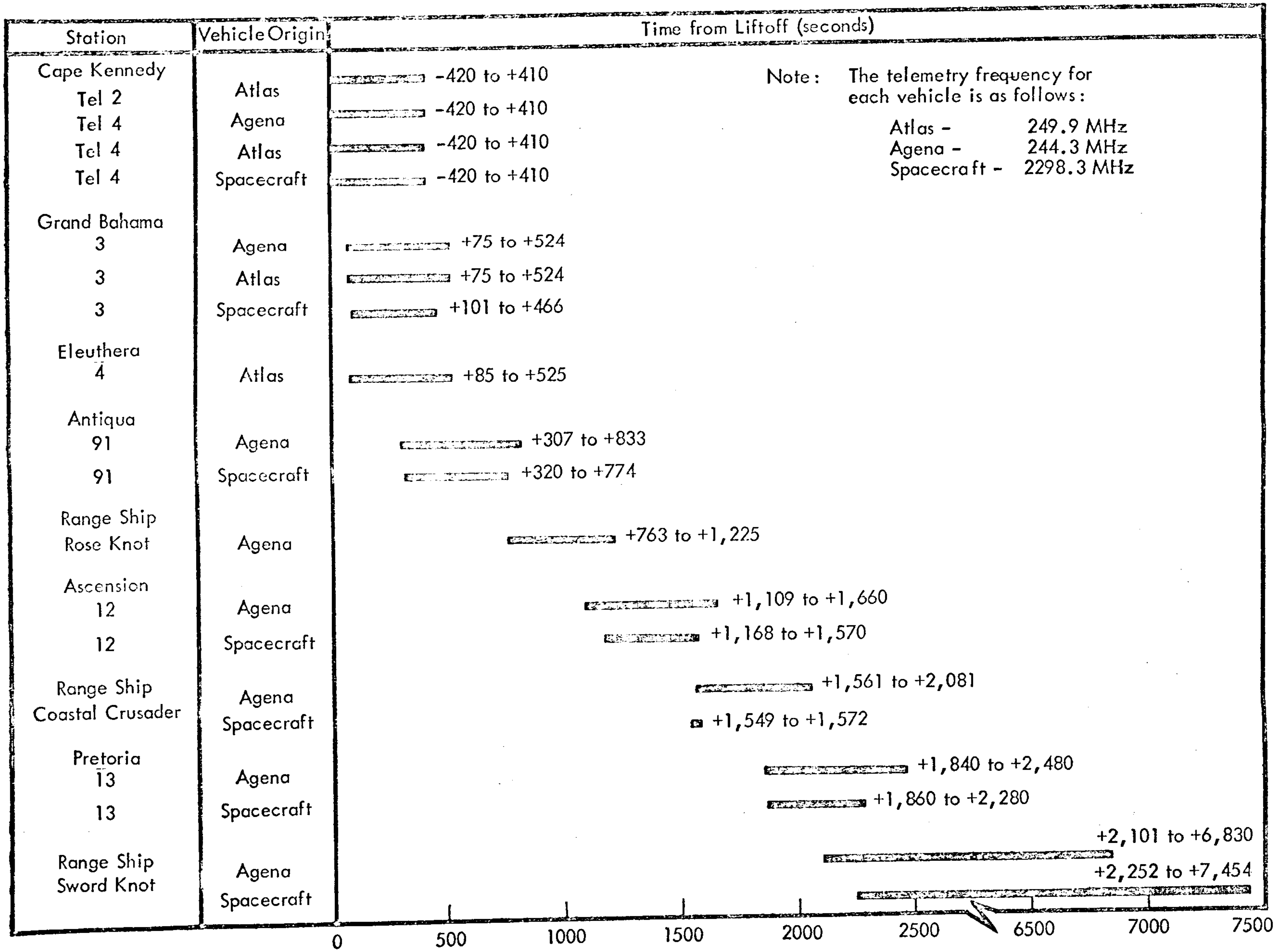


Figure 1-3: Telemetry Coverage

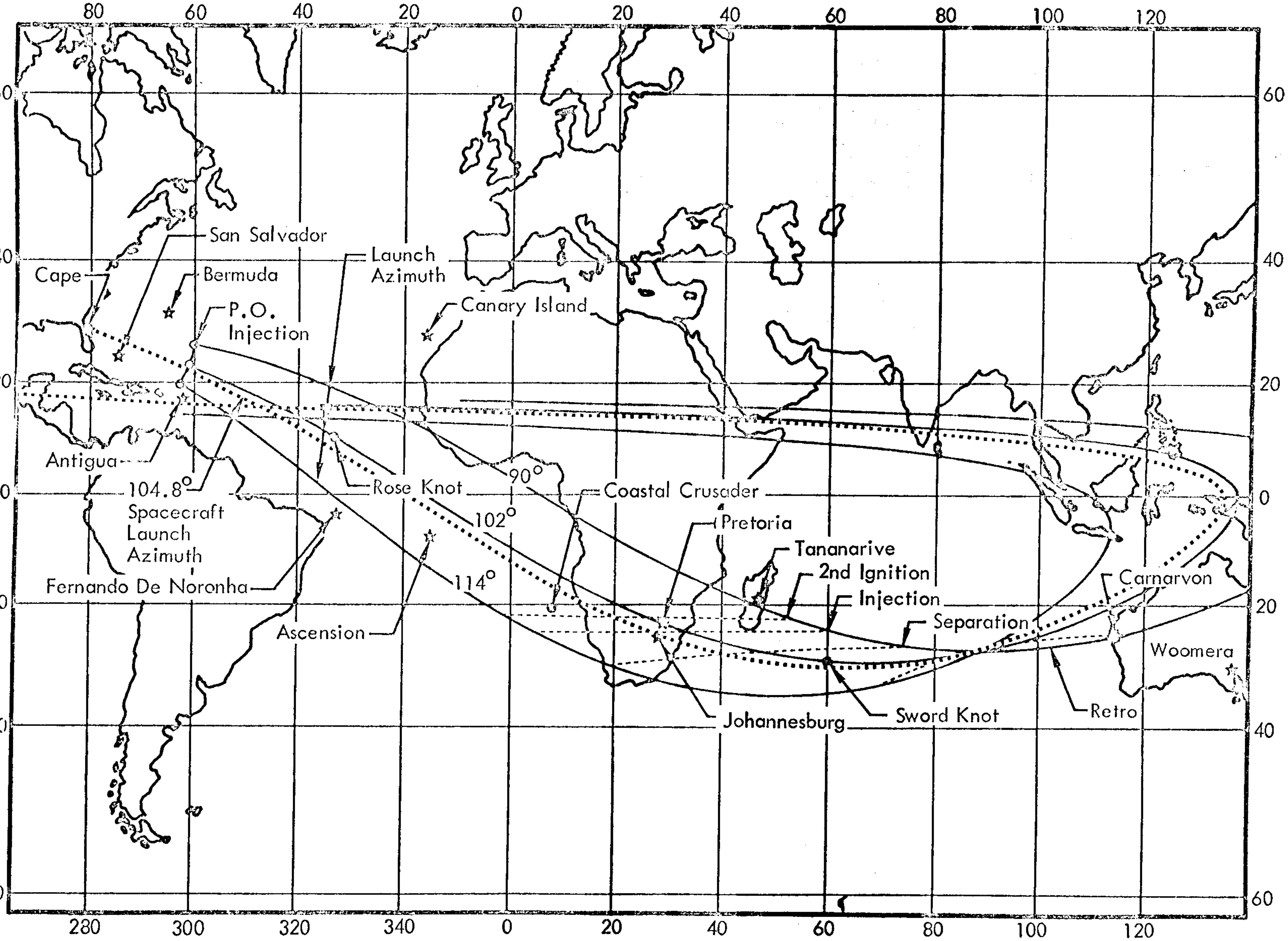


Figure 1-4: Ground Track for August 1, 1967

Table 1-6:
Sequence of Significant Flight Events

Event	Nominal Times	Actual Flight Times (sec)
Liftoff 22:33:00.352 GMT		-0-
Booster Engine Cutoff (BECO)	128.9	128.6
Booster Engine Staging	131.9	131.7
Start Agena Auxiliary Restart Timer	271.85	272.4
Sustainer Engine Cutoff (SECO)	287.9	288.6
Start Agena Primary Timer	291.8	296.3 (1)
Vernier Engine Cutoff (VECO)	308.1	307.9
Uncage Gyros	308.1	307.9
Jettison Horizon Sensor Fairings	308.1	307.9
Fire Shroud Ejection Squibs	310.5	310.3
Fire Atlas-Agena Separation Squibs	312.5	312.4
Initiate Agena First-Burn Sequence	364.8	369.3
Steady-State Thrust 90% Pc	365.95	370.45
Agena First-Burn Cutoff (VM)*	517.6	523.58
Stop Primary D-Timer		651.5
Restart Primary D-Timer		1,865.4
Initiate Agena Second-Burn Sequence		1,879.4
Steady-State Thrust 90% Pc	1,880.0	1,880.11
Agena Second-Burn Cutoff (VM)**	1,966.55	1,967.6
Fire Ejection Squibs (Spacecraft)	2,132.85	2,133.42
Initiate Yaw Maneuver	2,135.85	2,136.39
Stop Yaw Maneuver	2,195.85	2,196.27
Initiate Retrofire	2,732.85	2,733.5
Retrorocket Burnout	2,748.85	2,750.5

* First-Burn Duration 153.13 secs
** Second-Burn Duration 87.5 secs
Total Burn Time . . . 240.63 secs

(1) Primary D-Timer started four seconds late.

sion IV launch vehicle. The general space vehicle system configuration is shown in Figures 1-5, 1-6, and 1-7.

1.3.1 Atlas Performance

The Atlas launch vehicle had three primary objectives and one secondary objective in support of Lunar Orbiter Mission V. The primary goals were:

- Place the upper stage into proper coast ellipse;
- Initiate or relay commands properly for separation of the upper-stage vehicle and start the Agena primary timer;
- Relay commands to the Atlas-Agena interface to jettison the shroud and start the launch vehicle secondary timer.

The secondary objective was determination of Atlas performance by using telemetry data.

All objectives were successfully achieved. The 5-Hz longitudinal liftoff oscillation reached a maximum peak-to-peak amplitude of 1.15g at T + 4 seconds. This was the third time during the SLV-3 program that the 5-Hz liftoff oscillation reached an amplitude in excess of lg peak-to-peak. Amplitudes of this magnitude, however, were not considered detrimental to the vehicle structure. The oscillation was essentially damped out by T + 22 seconds. Telemetry data were satisfactory except for two data points. V2 chamber pressure decreased 10 psi at T+164 and at T +168.5 seconds and then abruptly increased back to its previous steady-state level. This data characteristic has been observed on previous SLV-3 flights and has been attributed to carbonization of the instrumentation sensing port. The second unsatisfactory reading was obtained from the engine fuel tank pressure and thrust chamber pressure which displayed data trends characteristic of transducer wiper-arm liftoff.

1.3.2 Agena Performance

The second-stage Agena vehicle had two primary objectives and one secondary objective in support of Lunar Orbiter Mission V. The primary goals were:

- To inject the spacecraft into a lunar coincident transfer (cislunar) trajectory within prescribed orbit dispersions.
- To perform Agena attitude maneuvers and retromaneuvers following Agena-spacecraft separation to ensure that the Agena would not, to the specified probabilities, intercept the spacecraft, pass within 20 degrees of the center of the Canopus tracker field of view, or impact the Moon.

The secondary aim of the Agena vehicle was to provide tracking and telemetry data for evaluation of Agena performance.

All objectives were satisfied. The only deviations from a nominal flight were the shroud separation monitor previously discussed and the fact that the primary sequence timer was started approximately 4 seconds late with all timer-controlled propulsion functions correspondingly late.

The Agena vehicle is in a long-lifetime Earth orbit with apogee and perigee altitudes of 369,831 and 9,380 kilometers, respectively. The period of this orbit is approximately 10 days. Comparison of the Agena parameters with those for the spacecraft shows that the retromaneuver was successful in ensuring that the Agena would not interfere with the spacecraft or its mission. The Agena missed the Moon by 25,317 kilometers at its closest approach.

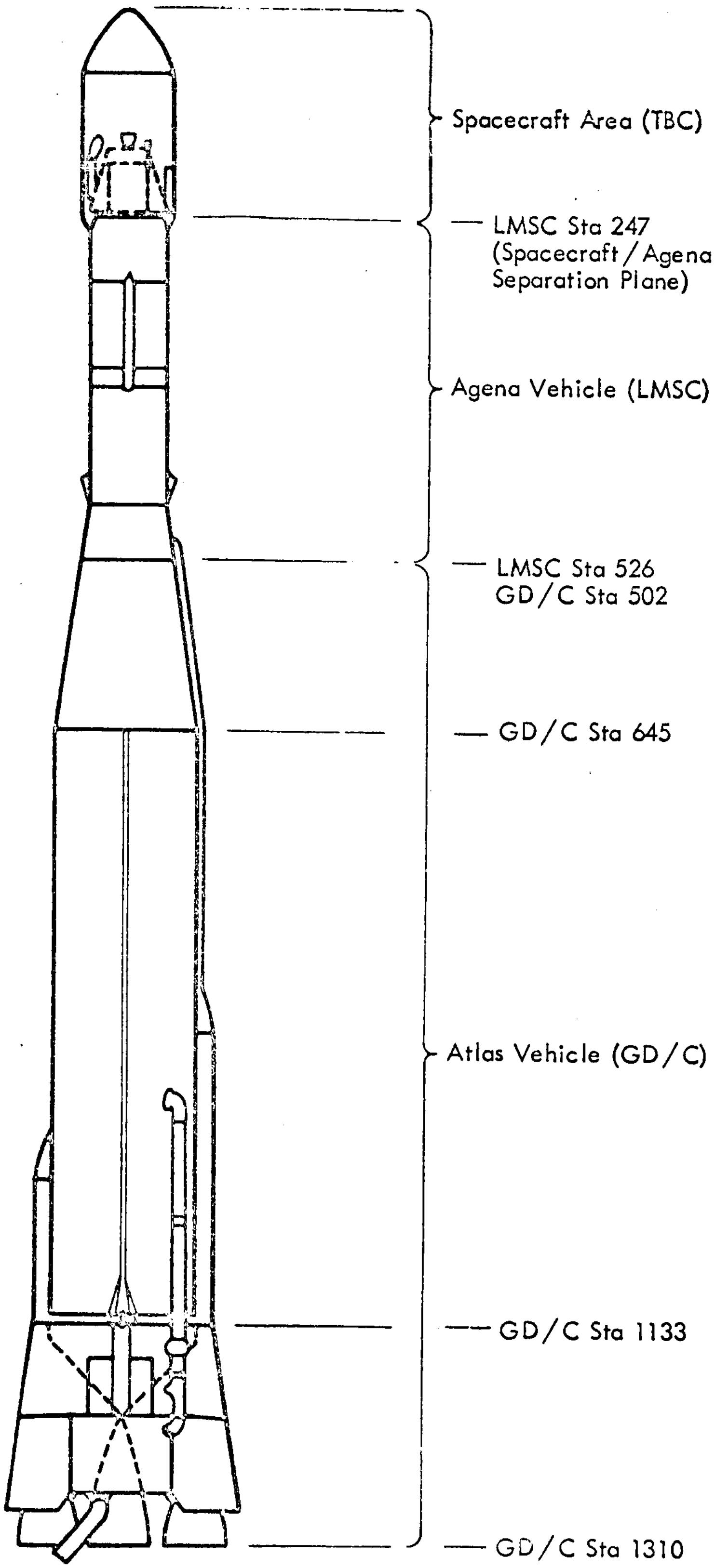
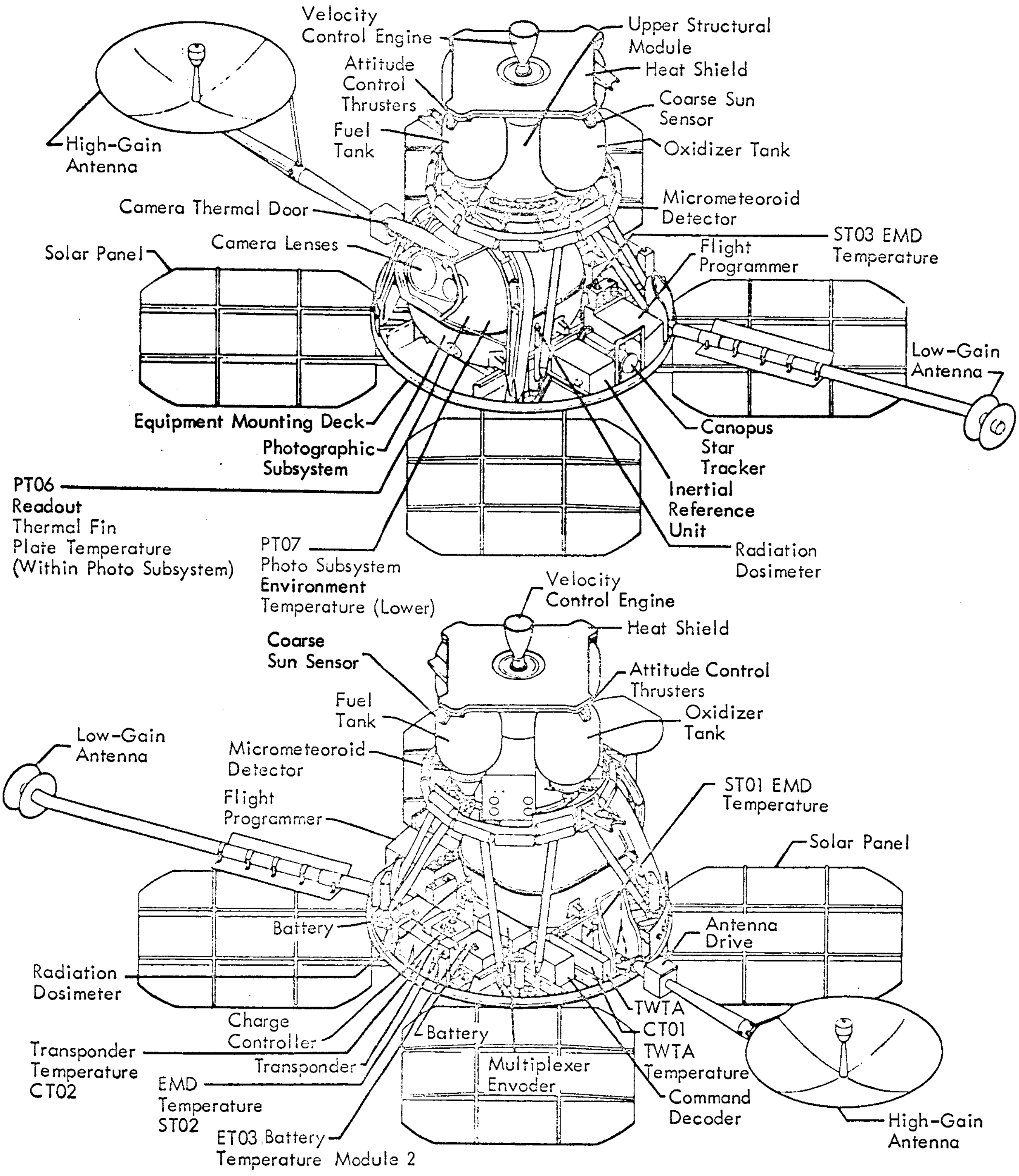


Figure 1-5: Lunar Orbiter Space Vehicle



NOTE: Shown with thermal barrier removed

Figure 1-6: Lunar Orbiter Spacecraft

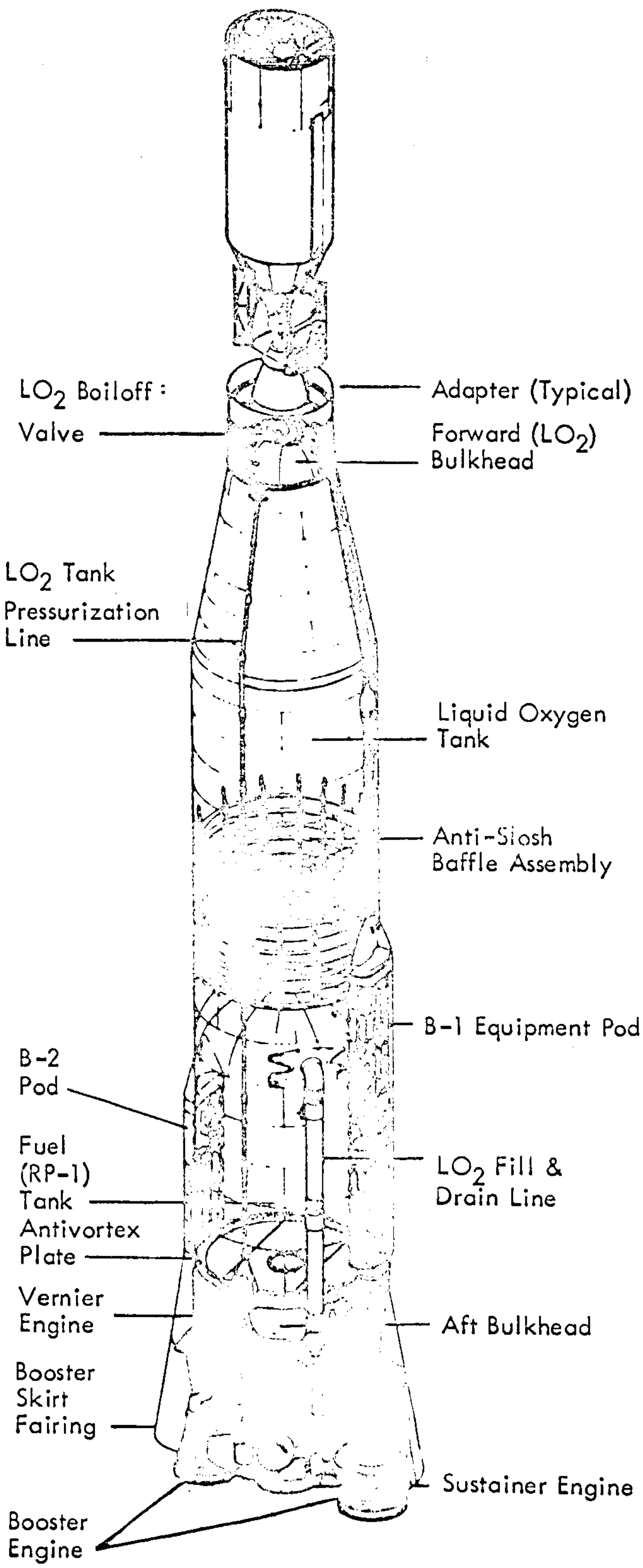


Figure 1-7: Launch Vehicle

2.0 Flight Operations

This section describes Lunar Orbiter Mission V flight operations from liftoff at Cape Kennedy, Florida, at 22:33 GMT, August 1, 1967, to the beginning of the extended mission at 02:00 GMT, August 28, 1967. Included are a comparison of the flight plan with the actual mission; a discussion of operational controls used to control spacecraft trajectory and performance; and descriptions of airborne and ground systems performance.

Mission V was nominal from liftoff through completion of final readout with one exception. During the "Bimat cut" sequence a "Bimat clear" indication was obtained 5 minutes prior to scheduled execution of the "Bimat cut" command. Investigation later revealed that the actual amount of Bimat was approximately 5 feet short of that normally loaded. During rewind of the Goldstone leader after completion of the readout, the leader broke, thus precluding any photo readout activity during the extended mission.

The flight operations team was essentially unchanged from that used on Mission IV. The high return rate of experienced personnel contributed in no small measure to the success of this highly complex mission.

Several areas of the Moon's farside were not photographed during Mission IV. The first eight Mission V orbits, during which photographic coverage of the farside of the Moon was completed, were similar to Mission IV. The remainder of the mission was accomplished at lower orbits similar to Missions I and III. The operational techniques developed in previous missions were used in conducting Mission V with little modification.

2.1 FLIGHT PLAN AND CONDUCT

This section describes the Lunar Orbiter Mission V flight plan and summarizes the nominal mission design. It should be noted that the mission was conducted with very little deviation from the mission design and flight plan.

2.1.1 Flight Plan

2.1.1.1 Mission Design

Design of the nominal mission, designated P-11A, required acquisition of 213 frames of photography from three different lunar ellipses. In this respect, the design differed from that of all previous Lunar Orbiter missions. To accomplish the desired photography, the spacecraft was to be injected into a 6,092-kilometer-apolune by 200-kilometer-perilune ellipse from which 17 frames covering areas of the lunar farside were to be exposed. In addition, two frames were required to satisfy the film-set constraint. A transfer maneuver was then programmed to reduce the perilune altitude to 100 kilometers for intermediate-ellipse photography. In this ellipse, six additional frames of farside photography were to be acquired; in addition, one frame was to be used as a film-set frame. A second transfer maneuver was then programmed to result in a final ellipse of 1,500-kilometer-apolune by 100-kilometer-perilune for the remaining photography of 45 selected nearside targets using the remaining 187 frames. Priority readout was to be conducted between photo exposures for mission control purposes according to a schedule designed to provide contiguous readout periods through Frame 38.

Table 2-1 contains a summary of primary parameters of the three ellipses planned for Lunar Orbiter V together with final-ellipse data for the other four Lunar Orbiter missions.

After exposure and processing of the 213 planned photos, a final readout of all photo data was planned.

2.1.1.2 Mission Description

The Lunar Orbiter V flight plan was based on the objective of photographing scientifically interesting sites on the nearside of the Moon. In addition, the latest mission design provided for 23 sites of farside photography to complete coverage of lunar areas not previously photographed.

Table 2-1: Primary Orbital Parameters

	L.O. V			L.O. IV	L.O. III	L.O. II	L.O. I
	Initial	Intermediate	Final				
Orbit period	8 hrs, 30 min, 29.1 sec	8 hrs, 22 min, 40 sec	3 hrs, 11 min, 14 sec	12 hours	3.5 hours	3.5 hours	3.5 hours
Orbit inclination (deg)	85.0	85.0	85.0	85.0	21	12	12
Perilune altitude (km)	200	100	100	2,700	60	60	60
Apolune altitude (km)	6,092	6,092	1,500	6,110	1,850	1,850	1,850
Time in sunlight (%)	100	100	100	100	75	75	75
Earth occultation (%)	2	0	15	0	29	29	29

The P-11A mission launch was planned for 22:03:03 GMT on August 1, 1967, with a launch azimuth of 102 degrees. A 90-hour cislunar trajectory was planned with midcourse corrections at 30 hours and 65 hours. A plane change of 13.79 degrees was planned for injection into lunar orbit. A waiting period of only 18 hours was planned between lunar injection and the first photo to be taken in the initial ellipse.

Priority readout was scheduled to be performed as necessary to achieve maximum data for precise mission control, even though some readout periods were only 4 or 5 minutes' duration. Including the time necessary to complete a final readout of all photo data, the total scheduled duration of the mission was 26.5 days.

2.1.1.3 Nominal Mission Trajectory and Orbital Parameters

For planning purposes, the nominal (P-11A) mission was designed based on a specific launch time within one of the three windows of the launch period of August 1 to 4, 1967. Significant trajectory and orbit data parameters planned for the P-11A mission follow.

Launch

- Launch date and time – Day 213 (August 1, 1967) 22:03:03 GMT
- Launch azimuth – 102 degrees
- Earth parking orbit coast time – 24 minutes, 16.0 seconds

Cislunar Trajectory

- Injection time – Day 213 (August 1, 1967), 22:37:27.9 GMT
- Injection location – 24.02°S, 29.41°E
- Transit time – 90.002 hours

Lunar Arrival

- Date and time of closest approach – Day 217 (August 5, 1967), 16:37:37 GMT
- Inclination of approach hyperbola – 86.14 degrees
- Perilune altitude of approach hyperbola – 666 km

Lunar Orbit Injection

- Injection time – Day 217 (August 5, 1967), 16:19:42 GMT
- Lunar location of injection – 83.50°S, 67.96°E
- Altitude of injection point – 1,269 km
- Plane change – 13.79 degrees
- ΔV – 644.4 meters per second

Initial Lunar Orbit Definition

- Apolune altitude – 6,092 km
- Perilune altitude – 200 km
- Inclination – 85.00 degrees
- Period – 8 hours, 30 minutes, 29.1 seconds
- Longitude of ascending node at injection – 118.12°E
- Argument of perilune at injection – 0.98 degree
- Longitude of Sun at injection – 180.90°E
- Sunlight – 100%
- Earth occultations – 2%

First-Orbit Transfer Maneuver

- First transfer date and time – Day 219 (August 7, 1967), 07:10:13 GMT
- Lunar location of transfer – 0.63°S, 83.19°W
- Altitude of transfer point – 6,090 km
- ΔV – 11.54 meters per second

Intermediate-Ellipse Definition

- Apolune altitude – 6,092 km
- Perilune altitude – 100 km
- Inclination – 85 degrees
- Period – 8 hours, 22 minutes, 40 seconds
- Longitude of ascending node at transfer – 96.75°E
- Argument of perilune at transfer – 0.74 degree
- Sunlight – 100%
- Earth occultations – none

Second-Orbit Transfer Maneuver

- Second transfer date and time – Day 221 (August 9, 1967), 05:15:21 GMT
- Lunar location of second transfer – 0.47°N, 71.45°E
- Altitude of transfer point – 99 km
- ΔV – 234.21 meters per second

Final-Ellipse Definition

- Apolune altitude – 1,500 km
- Perilune altitude – 100 km
- Inclination – 85 degrees
- Period – 3 hours, 11 minutes, 14 seconds
- Longitude of ascending node at transfer – 71.41°E
- Argument of perilune at transfer – 0.46 degree
- Sunlight – 100%
- Earth occultations – 15%

2.1.2 Flight Conduct

No significant flight plan deviations were required to complete Mission V as planned. Small changes to the flight plan were effected in real time, primarily to ensure adherence to the planned flight profile. Special workaround procedures were not required since significant spacecraft problems had not developed prior to completion of final photo data readout.

Throughout the mission, only one significant anomaly was experienced; this was after completion of the mission photographic phase when the leader parted approximately 7 feet from the splice during film rewind.

The following paragraphs summarize flight conduct in general terms. Additional details regarding flight parameters, spacecraft performance, and flight path control are included in later sections of this document. The times of significant mission events are summarized in Tables 2-2 and 2-3. The actual core map plan is in Table 2-4.

2.1.2.1 *Launch through Lunar Injection*

Prelaunch countdown was conducted with all items normal on Launch Plan 1-A until T-90. At this time a problem developed that required replacement of an Agena velocity meter. This was accomplished and checked out without delaying the count. At 18:33 GMT, August 1, a squall line approached Cape Kennedy, causing a hold in the countdown. After weather conditions improved, the count was resumed at 20:58 GMT using Launch Plan 1-K and a successful liftoff occurred at 22:33 GMT. The launch azimuth was 104.8 degrees.

The following summary is presented for convenience in comparing planned versus actual occurrence times of significant mission events.

Programmed spacecraft events occurred normally following liftoff. The antennas and solar panels deployed as planned, and shortly afterward the spacecraft was acquired by DSS-41. A pitch and yaw gyro drift test was performed to establish drift rates, which were determined to be well within design limits.

Canopus acquisition procedures were commenced at 05:21 GMT, August 2. The same glint (reflected sunlight) problems occurred that were experienced in previous Lunar Orbiter missions. Initial star mapping was inconclusive but a star was in the tracker field of view and was tracked to establish the roll drift rate. Arming and bleeding of the propellant lines was conducted and firing of the fuel and oxidizer squibs was accomplished. Initial Canopus track was obtained at 14:34 GMT, August 2, after two high-gain-antenna maps were made to help find and verify Canopus position.

Table 2-2: Synopsis of Significant Events

Event	Date	Planned Time (GMT)	Actual Time (GMT)
Launch	Aug. 1	22:03:03	22:33:00.3
Cislunar injection	Aug. 1	22:37:28	23:05:48
First midcourse maneuver	Aug. 3	04:00:00	06:00:00
Second midcourse maneuver	Aug. 4	15:00:00	Not Required
Lunar injection	Aug. 5	16:19:42	16:53:04
First photograph	Aug. 6	10:58:00	11:22:00
Transfer — intermediate ellipse	Aug. 7	07:10:13	08:43:48
Transfer — final ellipse	Aug. 9	05:15:21	05:11:05
Complete photography	Aug. 18	21:50:29	21:40:13
Complete final readout (100% of photo data)	Aug. 28	12:00:00 est.	238:01:10:00 (Aug. 26)

Table 2-3: Summary of Significant Events

Actual Time (GMT)	Event
213:22:33:00	Liftoff
213:22:35:8.1	Atlas booster engine cutoff (BECO)
213:22:35:11.8	Atlas booster jettison
213:22:37:32.4	Start Agena secondary timer
213:22:37:48.6	Atlas sustainer cutoff (SECO)
213:22:37:56.4	Start Agena primary timer
213:22:38:8.1	Atlas vernier cutoff (VECO)
213:22:38:10.1	Shroud separation
213:22:38:12.5	Atlas-Agena separation
213:22:39:10.4	Agena first ignition
213:22:41:43.8	Agena parking orbit injection
213:23:04:21.0	Agena second ignition
213:23:05:48.0	Agena cislunar injection
213:23:05:35	Agena yaw start
213:23:05:55	Agena-spacecraft separation
213:23:07:04	DSS-51 one-way
213:23:09:05	Agena yaw stop
213:23:10:32	Antenna deployment
213:23:10:57.8	Solar panel deployment
213:23:18:33	Agena retro fired
213:23:18:49.7	Agena retro stopped
213:23:18:56	DSS-41 one-way
213:23:23:00	Acquire Sun
213:23:23:17	DSS-41 two-way
213:23:25:00	Start gyro drift test

Table 2-3 (Continued)

Actual Time (GMT)	Event
213:23:32:00	Acquire Sun
214:01:12:00	Film supply cassette radiation dosage, DF04, incremented to 0.75 rad
214:04:20:00	Resume gyro drift test
214:05:00:00	Acquire Sun
214:05:21:00	Start Canopus search
214:18:34:00	Canopus located
214:18:39:00	Antenna map (run to verify Canopus location)
214:19:00:00	Canopus location verified
215:05:46:00	Start attitude maneuver for midcourse correction
215:06:00:00	Start engine for first midcourse
215:06:00:26.1	Engine burn complete; $\Delta V = 29.76$ meters/second
215:06:03:00	Start reverse maneuver
215:06:07:00	Acquire Sun
215:06:20:00	Gyro drift test
215:20:15:00	Start CTD test
216:03:10:00	Begin drift test
216:12:15:00	Acquire Sun
217:09:36:00	Begin drift test
217:12:30:00	Drift test complete
217:16:33:00	Start attitude maneuver of lunar injection
217:16:48:54.4	Start engine burn
217:16:57:12.5	Stop engine burn; $\Delta V = 643.0$ meters/second
217:16:59:00	Begin reverse maneuver

Table 2-3 (Continued)

Actual Time (GMT)	Event
217:17:04:00	Acquire Sun
217:18:24:00	Start Goldstone readout
217:18:44:00	Stop Goldstone readout
218:11:00:00	Advanced 12 frames
218:11:11:00	Mission V photography started
218:11:44:30	Film supply cassette radiation dosage, DF04, incremented from 0.75 to 1.00 rads
219:08:28:00	Begin attitude maneuver for transfer to intermediate orbit
219:08:43:48.4	Begin engine burn
219:08:43:59.2	Stop engine burn
219:08:45:00	Begin reverse maneuver
219:08:52:00	Acquire Sun
219:16:02:00	Mission V readout started
221:04:52	Begin attitude maneuver for transfer to final orbit
221:05:08:32.7	Begin engine burn
221:05:11:05.6	Stop engine burn
221:05:13:00	Begin reverse maneuver
221:05:21:00	Acquire Sun
230:21:40:13	Complete photography
231:03:19:00	Bimat-clear indication 5 minutes prior to execution of "Bimat cut" command
231:04:30:00	Final readout started
238:01:10:00	100% readout completed (all data received)
239:05:38:00	Final readout completed (film rewound)
240:02:00:00	Begin extended-mission operations

Table 2-4:
Core Map Plan Photographic Activity

Core Map	Orbit Loaded	Photo Orbit	Photo Sites [1]	Carryover from Previous Map
10	1	2	Film advance	—
		3	A-1, A-2	
10 update	1	3	A-3	—
11	3	4	A-4, filmset (A-5)	
			Orbit transfer	—
11 update	4	5	A-6	—
12	5	6	A-7	—
12 update	6	7	A-8	
12 update	6	8	A-9, A-10	—
		9		—
13	8	9	A-11	
13 update	8	10	A-12	
			Orbit transfer	
		11	A-13	
		12	—	
14	12	13	A-14	—
14 update	12	14	—	
		15	V-1	
15	15	16	—	
		17	V-2	—
		18		
16	17	19	V-3, A-15	

[1] V/H was on for all sites designated by V's

Table 2-4 (Continued)

Core Map	Orbit Loaded	Photo Orbit	Photo Sites [1]	Carryover from Previous Map
		20	V-4	
17	19	21	V-5.1	A-15, V-4
		22		
18	21	23	V-6	–
18 update	21	24	A-16	
		25	–	–
19	25	26	V-8a	
19 update	25	27	V-8b	
20	26	28	V-9	V-8b
20 update	27	29	A-17	
		29		
21	28	30	V-10	A-17
		31	V-11a	
22	30	32	V-11b	V-11a
22 update	30	33	V-12	
23	32	34	V-13, A-18	V-12
23 update	32	35	V-14	
24	34	35		A-18, V-14
		36	V-15.1	
		37	V-16a	
25	36	38	V-16b	V-16a
		39	A-19	

[1] V/H was on for all sites designated by V's

Table 2-4 (Continued)

Core Map	Orbit Loaded	Photo Orbit	Photo Sites [1]	Carryover from Previous Map
		40	–	
26	39	41	V-18	A-19
26 update	39	42	V-19	V-18, V-19
26	40	43	–	
27	42	44	A-20	
		45	V-21	
28	44	46	V-22	A-20, V-21
		47	V-23.1	
29	46	48	V-24	V-23.1
		49	V-25, A-21	
30	48	50	V-26	V-25, A-21
30 update	49	51	V-27a	
31	50	51		
		52	V-27b	
		53	V-28	
32	52	54	V-29, A-22	V-28
		55	V-30	
33	54	56	V-31	A-22, V-30
		57	V-32	
		58	–	–
34	57	59	V-33	
		60	V-34	

[1] V/H was on for all sites designated by V's

Table 2-4 (Continued)

Core Map	Orbit Loaded	Photo Orbit	Photo Sites [1]	Carryover from Previous Map
35	59	61	V-35	V-34
		62	V-36	
36	61	63	V-37	V-36
		64	A-23	
37	63	65	V-38	A-23
		66	–	
38	65	67	A-24	
		68	–	
39	67	69	V-40	A-24
39 update	67	70	V-41	V-40
40	69	71	V-42a	V-41
40 update	70	72	V-42b	
41	71	73	V-43.1	V-42b
		74	V-44	
		75		
42	75	76	V-45.1	
		77	V-46	
		78	–	
43	77	79	V-48	
		80	V-49	
		81	–	
44	80	82	V-50	
		83	V-51, Film advance	

[1] V/H was on for all sites designated by V's

A three-axis gyro drift test was conducted and accurate drift rates were determined. Spacecraft temperatures were carefully monitored; when temperatures increased, the camera heaters were turned off to avoid pitching off-Sun.

During the mission, Canopus was not "acquired" but was monitored by turning the Canopus tracker on for a few minutes every 3 hours, and by performing a roll update maneuver when necessary. This operational technique kept Canopus within the tracker field of view so that the roll error could be determined periodically. Also, by operating the tracker only a few minutes at a time, voltage degradation of the tracker was minimized.

The first midcourse maneuver was satisfactorily performed at 06:00 GMT, August 3, with a 26-second engine burn, which resulted in a ΔV of 29.76 meters per second. Throughout the remainder of the cislunar phase, the Canopus tracker was cycled for a few minutes every 3 hours. A test was performed to verify satisfactory operation of the camera thermal door. The very accurate performance of the first midcourse maneuver obviated the need for a second correction so the second midcourse maneuver was cancelled. Plans for the lunar injection maneuver were completed and the necessary commands loaded into the spacecraft flight programmer. Shortly before the scheduled deboost maneuver, the high-gain antenna was rotated in preparation for the Goldstone test film readout. Backup injection maneuver commands were also prepared and transmitted to the spacecraft, and alternate mission commands were prepared and sent to the DSS to be held in readiness.

The spacecraft was successfully placed in lunar orbit using a 498.1-second engine burn to produce a ΔV of 643 meters per second. The deboost burn was terminated at 16:57:12.5 GMT on August 5.

2.1.2.2 Initial and Intermediate Ellipses

The TWTA was turned on shortly after injection into lunar orbit, and at 18:24 GMT, August 5, the Goldstone test film readout was conducted. All parameters were normal except for unexplained momentary loss of video which caused random dropout of seven scan lines. Photography of the first photo site was conducted at 11:22 GMT, August 6, and subsequent photography was conducted according to plan. Film processing was commanded and all photo subsystem operations appeared to be normal (except for random scan line dropouts).

A total of four photo sites using 18 exposures was successfully completed as planned in the initial ellipse. Actual orbital parameters of this ellipse were perilune altitude, 202.0 kilometers; apolune altitude, 6,030.3 kilometers; period, 505 minutes; inclination, 85.1 degrees.

The first orbit transfer maneuver commands were prepared and loaded into the spacecraft. Photo data readout was not conducted in the initial ellipse.

After four initial orbits were completed, the first orbit transfer maneuver was performed at 08:44 GMT, August 7, using a 10.78-second engine burn to produce a ΔV of 15.97 meters per second. The actual orbital parameters of the intermediate ellipse were perilune altitude, 100.6 kilometers; apolune altitude, 6,064.3 kilometers; period, 501 minutes; and inclination, 84.6 degrees.

Intermediate-ellipse photography consisted of six lunar farside sites, each site being assigned one exposure. In addition, one of the film movement frames was assigned to a historic telephoto picture of the Earth. Periodic processing and two photo data readout operations were conducted in this phase according to the plan. All data continued to indicate good quality photographs, although sporadic scan line dropouts were to be noted in later readouts. These had no significant effect on photo quality.

After 5.5 orbits of the intermediate ellipse had been completed, a second transfer maneuver was successfully performed to place the spacecraft into the final ellipse. This maneuver employed a 152.9-second engine burn and produced a ΔV of 233.67 meters per second. This was the 24th consecutive successful rocket engine burn for Lunar Orbiter spacecraft.

2.1.2.3 Final-Ellipse Activities

Photographic Phase — Successful performance of the second transfer maneuver resulted in a final ellipse of 98.93-kilometer perilune altitude, 1,499.4-kilometer apolune altitude, 191.0-minute period, and 84.76-degree inclination.

The ambitious schedule of mission events as reflected in the flight plan increased the level of activities to a peak in the final-ellipse phase of the mission. This schedule was maintained successfully only because of the thoroughly experienced operations team and the absence of any significant system or spacecraft anomaly. Figure 2-1 shows a segment of four final orbits and depicts some of the numerous activities required to coordinate the effort necessary to obtain the desired photography.

Starting with Orbit 11, only 4.5 hours after the second transfer maneuver, site photography was conducted on almost every orbit of the final-ellipse phase until all 213 frames had been exposed. In addition to 13 apolune photos needed to complete the desired farside coverage, 184 frames were exposed to cover 45 scientifically interesting sites on the nearside of the Moon. All photos were successfully exposed, the film processing schedule was maintained, and a modified priority readout schedule was met. The readout schedule was modified during mission operations to eliminate readout of redundant data and those scheduled readout periods of less than 10-minute duration.

Photography of the last photo site, Site 51, was completed on Orbit 83 as planned. At this time it was decided to delay cutting the Bimat until after readout of some photo data of Sites 48 and 49. Accordingly, the processing and readout schedule of the plan was modified to obtain the additional priority readouts during Orbits 83 and 84. The Bimat-cut sequence was then rescheduled for Orbit 85. About 5 minutes before Bimat-cut was programmed to occur, a Bimat-clear signal was received via telemetry. Indication of Bimat exhaustion prior to the scheduled Bimat-cut was explained by a subsequent report that the supply of Bimat loaded prior to launch was about 5 feet short of that required. No particular problem resulted, except that Wide-Angle Frame 217 was not processed and Wide-Angle Frame 216 was degraded. No telephoto frames were lost.

Final Readout Phase — Final readout of all photo data was commenced in Orbit 85 and continued normally until the mission was completed. The schedule called for a readout each orbit of about 2.5 hours duration. As each readout was terminated, the spacecraft attitude reference was updated and the high-gain antenna was rotated to maintain optimum signal strength. Stored-program commands were loaded into the spacecraft flight programmer every fifth orbit.

Readout of 100% of the Mission V photo data was completed at 01:10 GMT, August 26 during Orbit 137. Final readout was completed in Orbit 146 on August 27 at 05:38 GMT with readout of High-Resolution Frame 5. Film rewind (readout without video transmission) was continued, however, to reposition all film on the film supply reel for permanent storage and to preserve readout capability. It was in the final stages of film rewind in Orbit 149 after final readout had been completed that the leader parted at approximately 7 feet from the splice, as noted previously.

During this mission, 213* frames were exposed and, with the exception of one wide-angle frame, all photo data were recovered. All 45 nearside sites and 23 farside areas were covered with good quality photos.

The spacecraft was configured for its extended mission during Orbit 152; at 02:00 GMT, August 28, the command programmer and all spacecraft subsystems were placed in the extended-mission configuration.

* Frame index numbers 5-217.

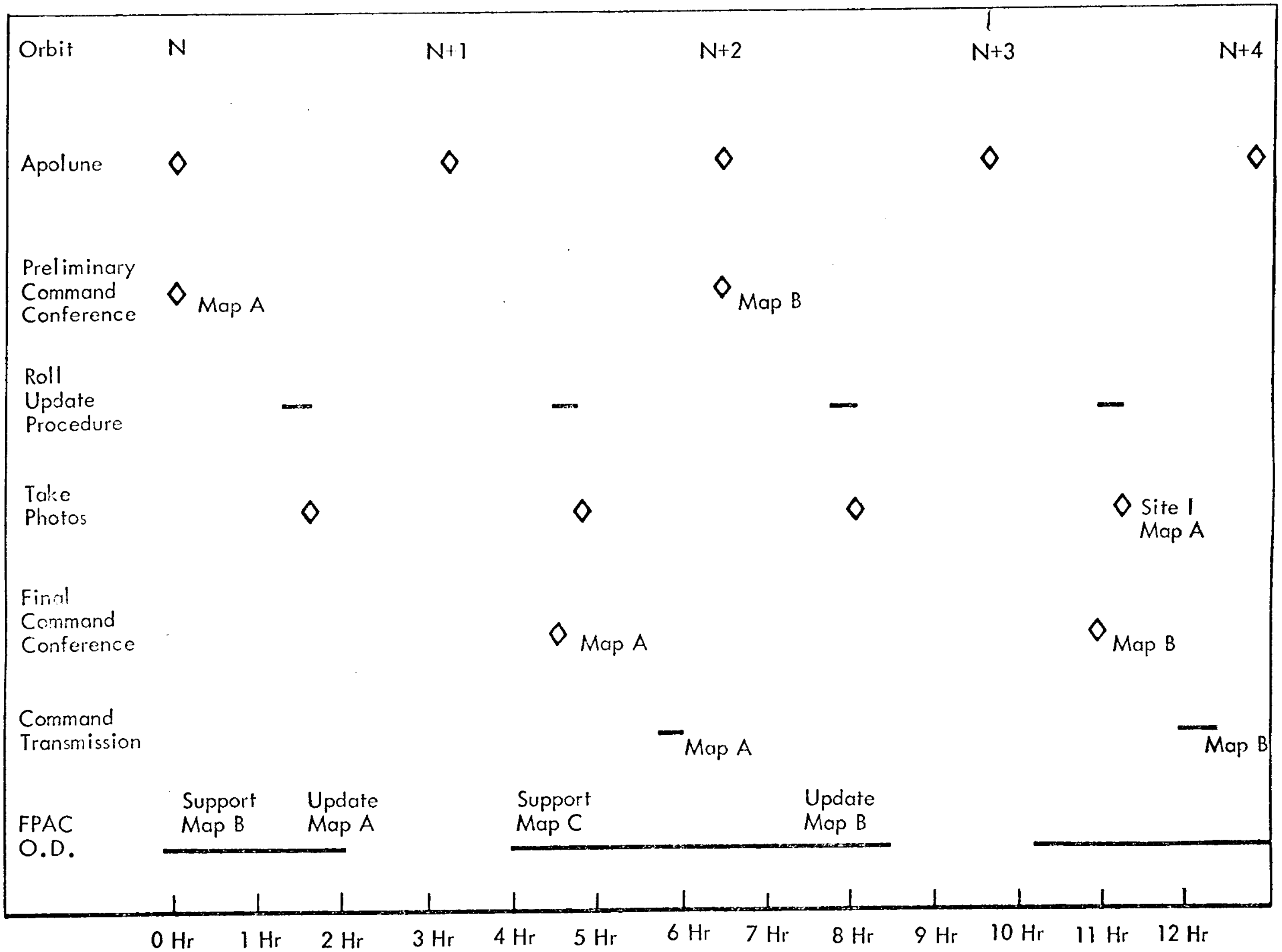


Figure 2-1: Typical Final-Ellipse Activity

2.2 FLIGHT CONTROL

2.2.1 Mission Control

Mission control activities are those required to integrate such operational areas as SPAC, FPAC, DSS, and data systems into a functional group that could successfully meet the objectives of the Mission V flight plan. On-line integration of these major operational areas was accomplished by the assistant space flight operations director (ASFOD). This position was actually staffed by two individuals, whose call names are ACE-2 and DEUCE-2, during all mission activities. Liaison between this position and the SPAC area was enhanced by the capability to observe closed-circuit TV monitors of selected 100-wpm printers, and the capability to communicate directly with the various subsystem analysts without using the main control network. Generally the spacecraft was flown within a few minutes of the schedule in the flight operations plan. However, operations directives provided a tool during the actual mission with which the flight operations plan could be rapidly modified in real time. This capability allowed the operations team to effectively meet mission objectives and, as minor problems occurred, to handle workaround techniques most expediently. For operational modifications or for tests to be performed on a "one-time" basis, operations orders were used again during Mission V.

The command coordinator position was occupied only by experienced personnel. Because of their thorough checkout and experience, no significant difficulties were encountered in spite of the complexity of the mission.

Mission event coordinator activities were performed similar to Mission IV in an efficient, trouble-free manner. Compared to the earlier missions, Mission V events adhered more closely to the flight plan, eliminating the need for extensive revisions to the sequence of events (SEAL program). In addition to the scheduled events, the Mission V sequence of events included the view periods and tracking schedules for Lunar Orbiters II and III, which were also in lunar orbit. The format for the teletype scripts that were sent to the DSS's remained the same. The use of SEAL charts given to each analyst was most effective on Mission V, primarily because of their increased accuracy, as well as the greater familiarity with the charts the users had developed through past experience. In all, 17 issues of the SEAL charts, showing several orbits' "scheduled" event times, were prepared during the mission and distributed to operations personnel.

2.2.2 Spacecraft Control

The following paragraphs describe the command programming and photo subsystem control procedures employed to meet the requirements of the Mission V flight plan (see Section 2.1). The personnel was substantially unchanged from Mission IV. This section discusses personnel activities involved in implementation of these controls.

2.2.2.1 Command Programming

As of August 27, at the end of the primary mission, a total of 4,525 commands had been prepared and transmitted to Lunar Orbiter V and properly executed by the flight programmer. Programming of commands followed the core map plan schedule in the flight operations plan. Major deviations from the premission plan were assignment of Photo Sites A-3 and A-9, which were originally reserved as filmset frames. Site A-3 covered a section of the moon farside while Site A-9 covered the earth taken from the moon. Despite the fact that this mission had the most demanding photographic assignment and the largest number of photo maneuvers, all command preparation activity was completed on a timely basis. This programming success can be attributed primarily to the absence of spacecraft anomalies and to a greatly improved schedule for providing updated FPAC command conference data to the command programmers. In addition, the retention of experienced command programmers contributed to meeting all programming requirements.

Premission Activity — Premission planning was much the same as for earlier missions. A core map plan defining the loading schedule of each map during photography and the photo orbit to be included in each map was prepared for overall coordination of command activity.

Countdown and Mode 2 commands for Mission V had been prepared and sent to the DSS near the end of Mission IV. Launch plans and initial core load commands were sent to DSS-71 during Mission V training.

Mission Activity — Command preparation was conducted by three teams of two command programmers each. Command programmer work schedules were adjusted such that a team attending the preliminary command conference for a core map would remain on shift through the final command conference for that map and complete all command preparation before handing over to the next team.

Preliminary command conferences convened 5.5 hours before scheduled command transmission. Frequently, the command conference data was given to the command programmers by FPAC well ahead of the conference. A COGL (command generation and program simulation) run with commands based on this preliminary data was made and the commands were then available for transmission to the spacecraft if final command conference data (command updates) were not provided on time. Command updates were provided by FPAC not later than 1 hour before the final command conference (2 hours before scheduled command transmission). This scheduling, a major improvement over earlier missions, enabled the command programmers to revise command sequences before the final command conference, thereby reducing the frequency of Mode 1 command updates transmitted to the spacecraft. On occasions when command updates were not obtained on time, the commands based on preliminary conference data were transmitted and later updated by a Mode 1 command update sequence to reflect the update. Additional changes to stored commands were made in Mode 3 for routine housekeeping tasks such as to compensate for roll axis reference or last-minute camera-on time changes.

2.2.2.2 *Photography Control*

Exposure and photo subsystem control procedures were basically the same as those used in Mission III. Site photography followed the mission plan with only a few minor changes.

One scheduled film-set frame (A-3) was used for a farside photo and another (A-9) was used for an Earth photo. Minor changes in target locations were made for five nearside and four farside sites.

Exposure Control — Exposure control for Mission V photography was accomplished — as for previous missions — by careful selection of shutter speeds. Recommended albedos and albedo distributions were again supplied by the U.S.G.S. for each nearside site. These albedo values were reduced by a factor of 1.3 for input to the photo quality prediction program (QUAL). Spacecraft film density predictions obtained from QUAL were the prime data for making shutter speed decisions. The H and D curve used for QUAL appears in Volume II of this series of documents. Although the loss due to the folding mirror in the 610-mm camera was not initially entered in QUAL, no decisions were affected up to the time it was changed. In marginal cases, discussions with advisors from U.S.G.S. and NASA resolved these decisions by considering the features of interest and type of terrain.

The farside photos were all taken at a shutter speed of 0.04 second due to their proximity to the terminator and the resultant low lighting conditions.

The availability of a "nominal book" and the fact that the mission was very close to nominal were of assistance in looking ahead during the mission to determine potentially marginal cases.

Camera-On Time Bias — For "V/H off" photography, the standard -0.9-second bias for camera-on time was used to correct for normal camera actuation delays. As data were received from priority readout, comparison of actual versus predicted exposure times indicated a bias of -1.0 second would result in a more exact exposure time. This change was implemented starting with Site V-25.

For "V/H on" photography, a bias was introduced to obtain a uniform distribution of actual exposure times about the desired times. The bias was adjusted to minimize the camera-on time uncertainty caused by the V/H cycle, and calculated using the predicted V/H ratio in the following equation:

$$\text{camera-on bias} = -0.050/(V/H)$$

Observed deviations from scheduled times are recorded in Volume II of this document series.

As photos were received during priority readout, a comparison of predicted versus actual photo coverage indicated a significant downtrack error. To compensate for this apparent error, a -3.0 second bias was introduced beginning with site V-26.1. This -3.0 seconds was added to the bias required by the photo-subsystem operation to obtain a corrected camera-on time for the flight programmer.

2.2.3 Flight Path Control

From launch through completion of photographic readout, maintaining control of the spacecraft trajectory (or flight path) was the responsibility of Flight Path Analysis and Command (FPAC). Responsibility for control of the mission from prelaunch checkout through about launch plus 6 hours belonged to the DSN FPAC. After the spacecraft had been acquired and was supplying good tracking data to the SFOF, the DSN FPAC team was relieved by the project FPAC team. At this point the project FPAC team assumed responsibility for flight path control for the remainder of the mission. Within both teams the tracking data analysis function was carried out by a JPL analyst. A description of the two FPAC teams is contained in NASA Document *Lunar Orbiter I Final Report – Photographic Mission Summary,* Volume I.

Flight path control by the FPAC team entailed execution of the following functions:

Tracking Data Analysis – (1) Monitoring and passing judgement on the quality of the incoming radar tracking data (doppler and range), which is the sole link between the spacecraft and FPAC, and the basis for determination of the current position and velocity of the vehicle; (2) Preparation of tracking predicts to support the DSS in spacecraft tracking.

Orbit Determination – A process of finding a trajectory that "best fits" the tracking data, which includes the tasks of editing the raw tracking data into a form acceptable to the orbit determination computer program (ODP), and subsequent operation of this program to obtain that trajectory best fitting the data – usually a lengthy task that consumes large blocks of computer time.

Flight Path Control – When the orbit determination process yields a trajectory, the flight path control function is initiated to determine the need for a corrective maneuver or the design of a planned maneuver. Thus, this function is principally one of guidance, control and prediction.

FPAC executes these functions to design maneuvers which will best achieve the objectives of the nominal flight plan. The computer programs, or FPAC software system, used for maneuver designs were identical to that used during Mission I with the exception of some internal modifications to individual programs. A description of the FPAC software system is contained in NASA Document CR 66324 *Lunar Orbiter I Final Report – Mission System Performance.*

From a trajectory point of view, the mission can be subdivided into the following phases:

Countdown, Launch, and Acquisition Phase – Covers the period from FPAC entry into the countdown through DSN acquisition of the spacecraft and subsequent handover from DSN FPAC team to Project FPAC team.

Injection through Midcourse – From completion of the second Agena burn through completion of the midcourse maneuver. This phase overlaps the acquisition portion of the previous phase.

Midcourse through Deboost – From end of midcourse burn through completion of the deboost maneuver.

Initial Ellipse — From end of deboost burn until the first transfer maneuver.

Intermediate Ellipse — From end of first transfer burn until the second transfer maneuver.

Final Ellipse — From end of second transfer burn until completion of readout

Table 2-5 lists principal FPAC events and their times of occurrence (GMT) within these phases. The orbit determination and flight path control functions executed in these phases will be discussed in the following subsections.

2.2.3.1 Countdown, Launch, and Acquisition
Performance by the DSN during countdown, launch, and acquisition in terms of the support required at the SFOF and the RTCS (real-time computer support) facility are indicated below.

SFOF — SFOF computer support requirements included:

- Running DSN and project prelaunch checkout cases;
- Processing ETR and MSFN static points;
- Running expected and actual liftoff time predict cases;
- Processing MSFN, ETR, and DSN raw tracking data, and determining orbits based on this data (Orbits 1101, 1103, 1105, and 1107).
- Running DSN predicts based on these orbits.

All these items were run in accordance with the Lunar Orbiter Network countdown. The Lunar Orbiter DSN FPAC team was not able to determine an orbit based on Pretoria (13.16) tracking data since only six points of good data were available (Orbit 1101). All other items were completed successfully.

RTCS — RTCS requirements included:

- Reformat MSFN and ETR static points;
- Reformat MSFN and ETR raw C-band radar tracking data;
- Process MSFN and ETR tracking data to determine parking orbit elements, injection conditions, and interrange vector (IRV);
- Theoretical transfer orbit and IRV based on the parking orbit and nominal Agena second-burn performance;
- DSIF predicts for DSS-51 and -41 based on parking orbit and nominal Agena second-burn performance;
- Processing MSFN and ETR tracking data to determine the first actual transfer orbit elements and injection conditions and IRV (first ETR orbit);
- DSIF predicts for DSS-51 and -41 based on the actual transfer orbit;
- Map the first actual transfer orbit to lunar encounter;
- Process DSN tracking data to determine a second actual transfer orbit (second ETR orbit);
- Map the second actual transfer orbit to lunar encounter.

All these items were successfully run on time by the RTCS facility, plus a third actual transfer orbit based on DSN tracking data which was also mapped to lunar encounter (third ETR orbit).

Other Items of Support

DSS 71 — This station provided prelaunch frequency reports at L-80, L-30, and L-6 minutes, and frequency parameters were provided to the RTCS at L-60 and L-20 minutes for use in DSIF predict generation. An updated frequency report was also received by voice from DSS-71 during the hold period.

Mark Times — Nominal prelaunch mark times were received from the launch operations center for both the 1A and 1K launch plans. The actual mark time reports were received from the MSFN and ETR telemetry stations (see Table 2-6).

Flight Path Analysis — Mark times and all trajectory data received from ETR and generated at the SFOF were evaluated in real time to determine mission status. This evaluation was performed at the Mission Operations Center in Hangar AO at ETR and at the SFOF.

Table 2-5: Principal FPAC Events

Launch and Acquisition	
• Aug. 1, 16:00	FPAC begins prelaunch checkout of software system
• Aug. 1, 22:33	Launch
• Aug. 1, 22:42	Agena first burn complete. Start 578-sec. coast
• Aug. 2, 23:06	Agena second burn complete. Cislunar injection
• Aug. 2, 23:40	First DSS-41 two-way doppler data
• Aug. 2, 02:00	DSN FPAC hands over control to Project FPAC
Injection through Midcourse	
• Aug. 2, 04:00	Discovered no necessity for early midcourse correction
• Aug. 2, 07:15	Midcourse maneuver time of August 3, 06:00 GMT selected
• Aug. 2, 19:30	Calculated 30 m/sec midcourse ΔV
• Aug. 3, 06:00	Start midcourse burn
Midcourse through Deboost	
• Aug. 3, 11:20	Determined second midcourse not required
• Aug. 5, 04:00	Completed design of deboost maneuver
• Aug. 5, 16:49	Start injection burn
Initial Ellipse	
• Aug. 5, 18:52	Obtain first post deboost orbit determination (OD 4102)
• Aug. 6, 11:22	Start of photography
• Aug. 7, 03:00	Completed design of first transfer maneuver
• Aug. 7, 08:44	Start first transfer burn
Intermediate Ellipse	
• Aug. 7, 10:36	Obtained first posttransfer orbit determination (OD 5302)
• Aug. 8, 12:00	Completed design of second transfer maneuver
• Aug. 9, 05:09	Start second transfer burn
Final Ellipse	
• Aug. 9, 07:31	Obtained first posttransfer orbit determination (OD 6302)
• Aug. 18, 21:40	End of photography
• Aug. 27, 05:38	Termination of readout (film rewound)

Table 2-6: Powered Flight Trajectory Events		
Mark	Event	Actual Time (GMT)
0	Liftoff	Aug. 1, 22:33:00:00
1	Atlas booster engine cutoff (BECO)	22:35:08.90
2	Atlas booster engine jettison	22:35:11.76
3	Start Agena secondary timer	22:37:32.36
4	Atlas sustainer engine cutoff (SECO)	22:37:48.56
5	Start Agena primary timer	22:37:56.36
6	Atlas vernier engine cutoff (VECO)	22:38:08.06
7	Shroud separation	22:38:10.26
8	Atlas-Agena separation	22:38:12.46
9	Agena first ignition	22:39:10.36
10	Agena shutdown (parking orbit injection)	22:41:43.46
11	Agena second ignition	23:04:20.66
12	Agena second shutdown (cislunar injection)	23:05:47.66
13	Agena-spacecraft separation	23:08:33.46
14	Begin Agena yaw	23:08:36.56
15	End Agena yaw	23:09:36:56
16	Agena retro	23:18:33.46
17	Stop retro	23:18:49.66

2.2.3.2 *Injection Through Midcourse Orbit Determination* – The following table shows the chronological sequence of the lunar encounter parameters obtained from the seven Project orbit determinations performed before midcourse.

Orbit Solution	Solution Type†	$\bar{B}\cdot\bar{T}$ (km)	$\bar{B}\cdot\bar{R}$ (km)	Time of Closest Approach (GMT Day 217)	Data Arc Length (hrs:min)
1202	B	6,128.2	3,534.6	18:25	00:27
1304	A	6,928.6	3,631.6	18:29:05.9	01:48
1306	A	6,888.2	3,479.5	18:28: 3.92	06:06
1108*	A	6,881.4	3,473.8	18:28:01.29	09:51
1210**	A	6,883.0	3,481.2	18:28:04.6	16:39
1212***	A	6,883.0	3,482.4	18:28:05.8	20:30
1314	A	(invalid encounter conditions obtained)			26:30
1099	A	6,887.7	3,478.2	18:28:06.2	30:18

* Used for preliminary command conference.
** Used for final command conference.
*** Used to check final command conference.
† Type A is two-way doppler data only. Type B is two- and three-way doppler and angle data.

Final design of the midcourse maneuver used OD 1210, which was based on 16 hours, 39 minutes of two-way lock doppler data from DSS's-12, -41, and -62. Doppler data fit by the orbit estimate was excellent for all three stations. Mark 1A range unit data and three-way doppler data were available in this data span but were used only as a check on the solution and were not allowed to influence the solution. The range unit residuals were small (on the order of 60 meters and smaller) and the three-way doppler residuals were very constant at the expected doppler bias levels.

A "best estimate" determination (OD 1099) done after the midcourse maneuver using all data up to the maneuver, indicated that OD 1210 predicted $\bar{B}\cdot\bar{T}$ within 4.7 km, $\bar{B}\cdot\bar{R}$ within 3.0 km, and time of closest approach (TCA) within 1.6 seconds. (One-sigma uncertainty in $\bar{B}\cdot\bar{T}$ was 11.0 km, $\bar{B}\cdot\bar{R}$ was 15.4 km, and TCA was 7.1 seconds for OD 1210.)

Appendix B, Vol. VI of this document series contains the inflight orbit determination summary reports. Figure 2-2 shows the aim point dispersions in $\bar{B}\cdot\bar{T}$, $\bar{B}\cdot\bar{R}$, and TCA.

Midcourse Design and Execution – Within 2 hours after cislunar injection, projected lunar encounter parameters (see Figure 2-2) indicated that the second Agena burn had resulted in a trajectory well within the midcourse capability of the spacecraft. It also became apparent that, although a midcourse maneuver would be required, the midcourse execution time would not be critical and an early midcourse would not be necessary.

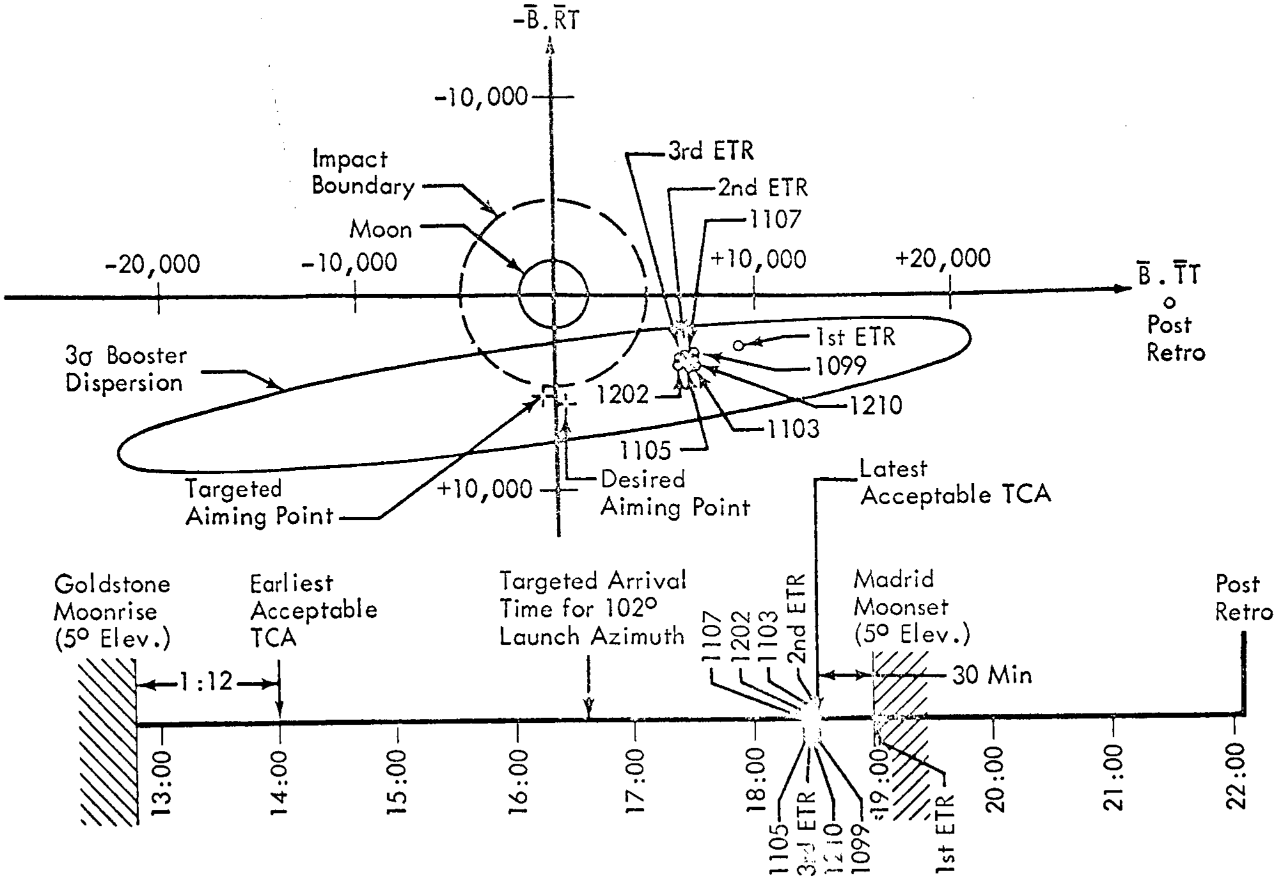


Figure 2-2: 3 σ Arrival Time Dispersion

The guidelines normally used in considering possible midcourse maneuvers were:

- Delay the maneuver as long as practicable to minimize the effect of midcourse execution errors on lunar encounter conditions;
- Perform the first midcourse maneuver at least 50 hours before orbit injection to allow time for a second midcourse;
- Minimize the Δ V required for lunar ellipse injection (deboost), transfer, and midcourse with a maneuver at the selected midcourse time.

OD 1304, based on 1.8 hours of tracking data, became available within 5 hours after cislunar injection. This OD solution was used for a study of midcourse execution time, correcting both the time of flight to the nominal encounter time (August 5, 17:10 GMT), and the miss parameters ($\bar{B}\cdot\bar{T}$ and $\bar{B}\cdot\bar{R}$) to those computed in the midcourse targeting program. Figure 2-3 shows the results of this study. The midcourse maneuver could have been delayed until 40 hours after cislunar injection without requiring excessive Δ V for the maneuver. Optimization of deboost Δ V is done automatically by the FPAC software programs for a given midcourse execution time and a specified lunar encounter time. By varying the arrival time for a selected midcourse execution time, it is possible to select the arrival time yielding lowest Δ V among these solutions. The results of this analysis are shown in Figure 2-4 for a midcourse executed 30 hours after cislunar injection.

Design of the photographic coverage for Mission V was indexed to the coverage of Site V-8a, which required that the spacecraft be at a specified latitude and longitude at a specified time regardless of arrival conditions. (Ordinarily, a change to arrival time would also affect each

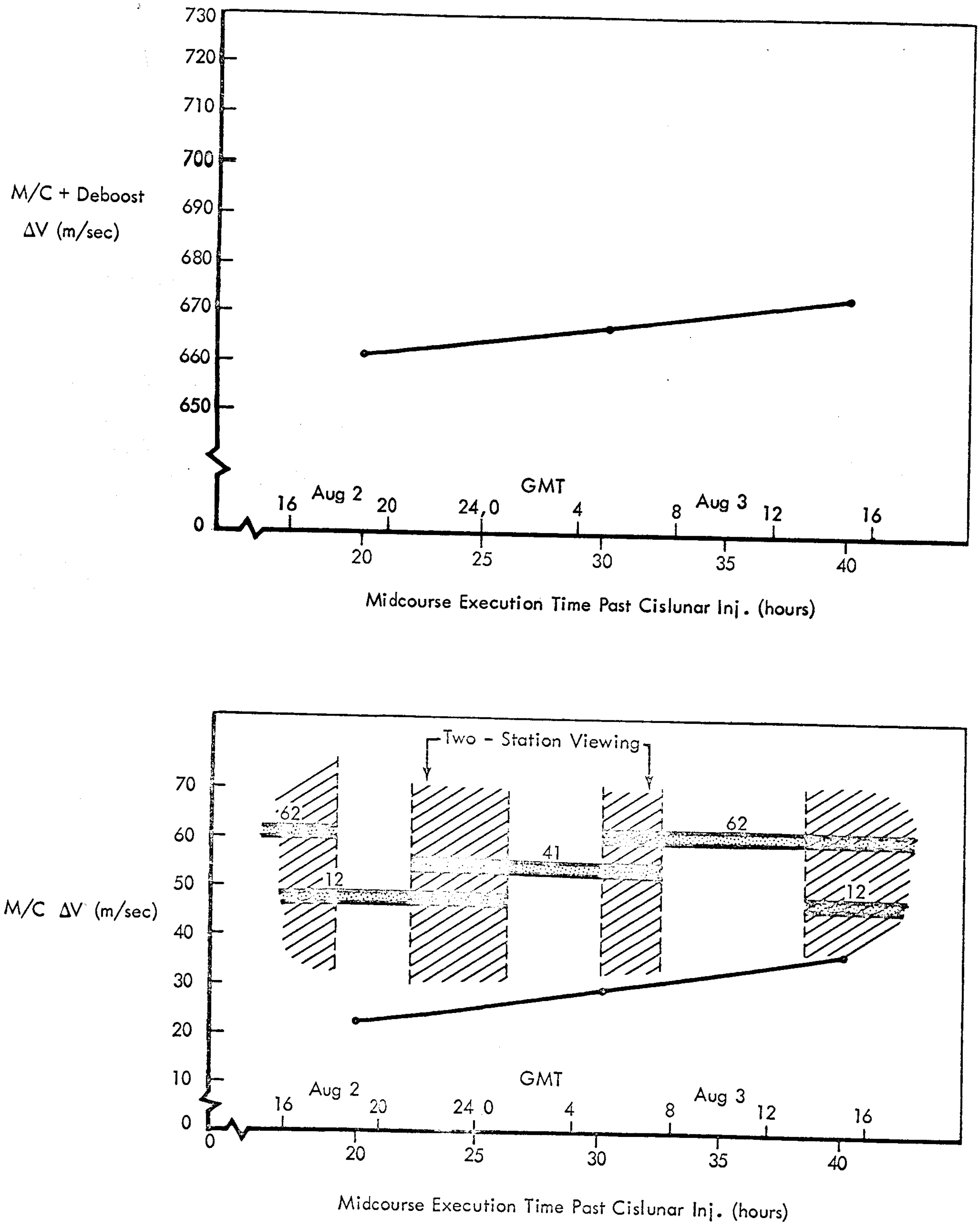


Figure 2-3: ΔV vs Midcourse Execution Time

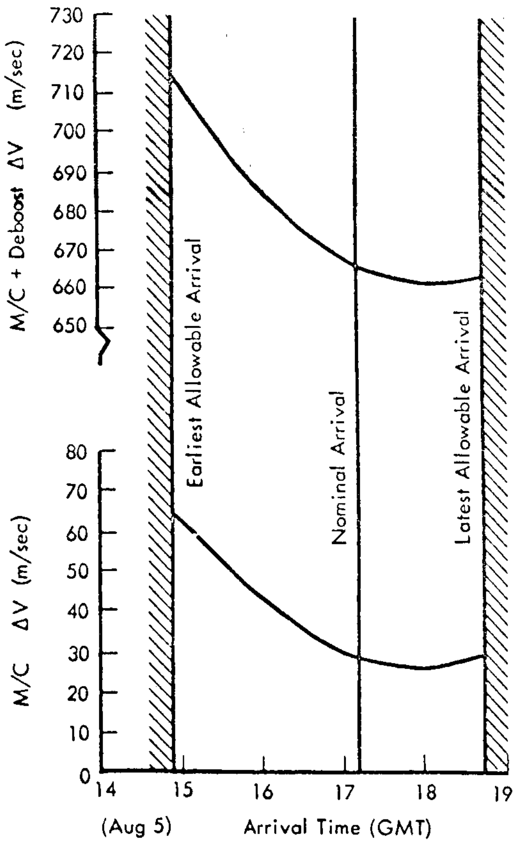


Figure 2-4: Δ V vs Arrival Time

photo time.) Thus, as arrival time was varied in the midcourse design, initial orbit period – via apolune altitude – was also varied enough to result in identical spacecraft ground traces in final orbit.

The minimum Δ V in this case could have been obtained by correcting the arrival time to August 5, 18:20 GMT. The desire for fuel conservation, however, was overridden by the need for some additional two-station viewing time after deboost. Therefore, an arrival time corresponding to the nominal preflight plan, 17:10 GMT, was arbitrarily selected. Midcourse execution time, August 3, 06:00 GMT, was chosen on the basis of desirable two-station viewing during and after the burn. DSS-62 viewing began 50 minutes before engine ignition to overlap with DSS-41. Two backup midcourse maneuvers were also designed against a possible abort situation. One was 5 hours later, during DSS-62 only viewing, the other 8 hours later, after DSS-12 rose, in case 62 became inoperative.

The midcourse maneuver specified by FPAC was:

- sunline roll, 42.07 degrees;
- pitch, 29.09 degrees;
- Δ V, 29.76 m/sec.
- Ignition time, August 3, 06:00:00 GMT

This attitude maneuver was selected from among 12 possible two-axis maneuvers on the basis of (1) maintaining sun lock as long as possible, (2) viewing DSS line-of-sight vector not passing through any antenna null regions, and (3) minimizing total angular rotation. OD 1210 was used for the midcourse final design.

Midcourse targeting resulted in the following set of encounter parameters, which are presented graphically in Figure 2-5.

	Nominal (Preflight Design)	Pre-Midcourse (Actual)	Post-Midcourse (Maneuver Design)
$\overline{B} \cdot \overline{T}$ (km)	380	6,888	385
$\overline{B} \cdot \overline{R}$ (km)	5,700	3,478	5,701
TCA (GMT)	August 5, 17:10	August 5, 18:28	August 5, 17:10
Δ V (km/sec)	0.935	0.9183	0.9396

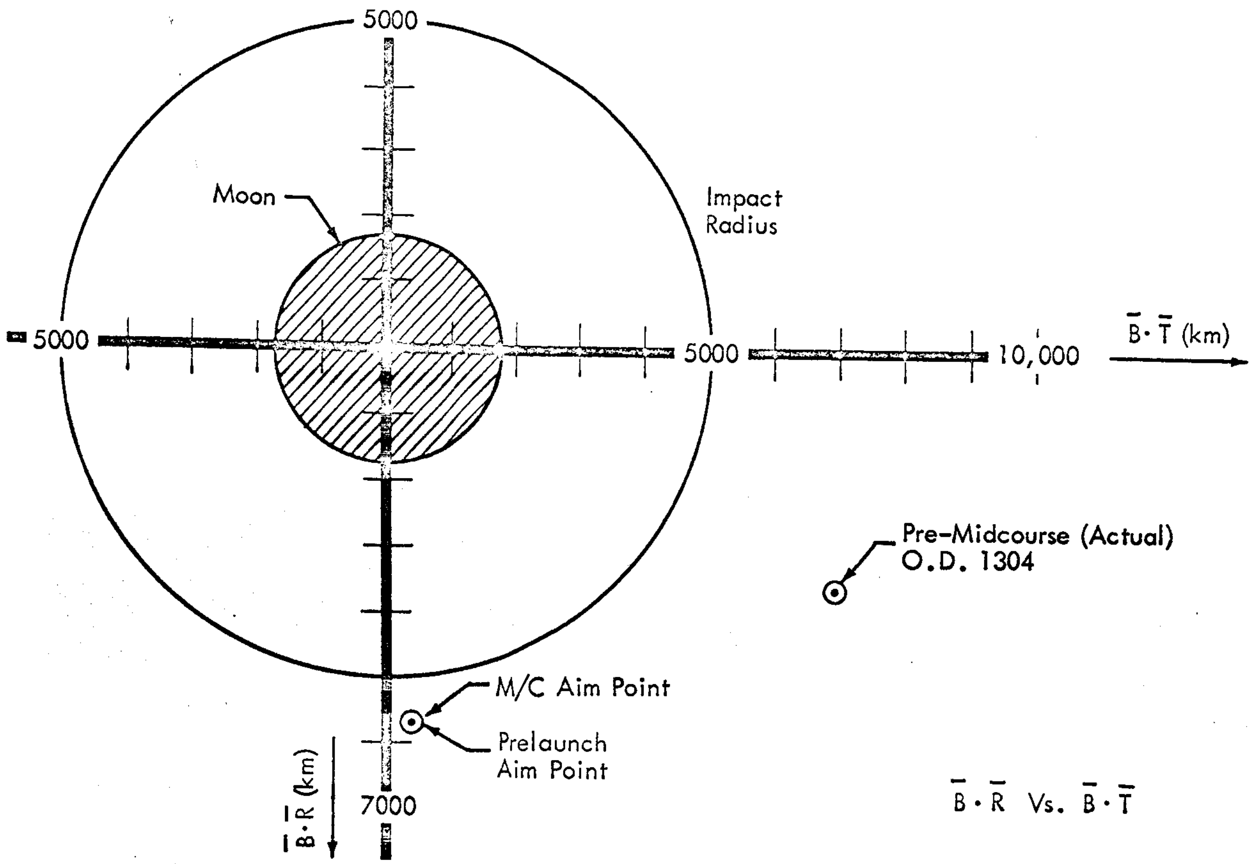


Figure 2-5: Pre-midcourse Encounter Parameter Summary

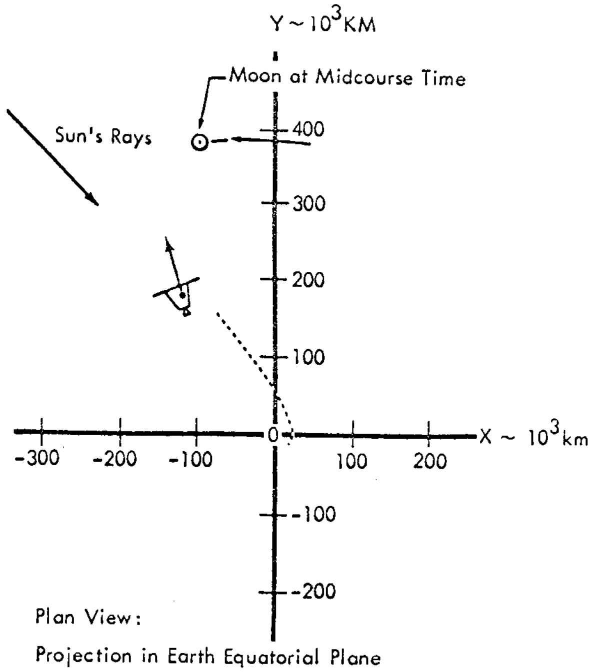


Figure 2-6: Midcourse Geometry

Figure 2-6 shows Earth-Moon-spacecraft geometry at the time of the midcourse maneuver and the direction of the desired velocity change. Engine ignition occurred at August 3, 06:00 (GMT) and the engine burned for 26 seconds, resulting in a doppler shift of 430 Hz. The doppler data observed during the burn indicated a nominal burn as shown in Figure 2-7.

Doppler Data Monitoring during Midcourse Maneuver — Several hours before the midcourse maneuver, a set of engine burn doppler predicts was computed that used the OD 1212 state vector, and nominal maneuver angles and engine performance parameters to predict doppler shift frequencies before, during, and after the maneuver. These predicted doppler frequencies were then plotted in the region of the burn and the actual doppler shift data plotted on the same curve as it was received in the SFOF. The resulting plot, Figure 2-7, shows that the maneuver was nominal to the accuracy of the

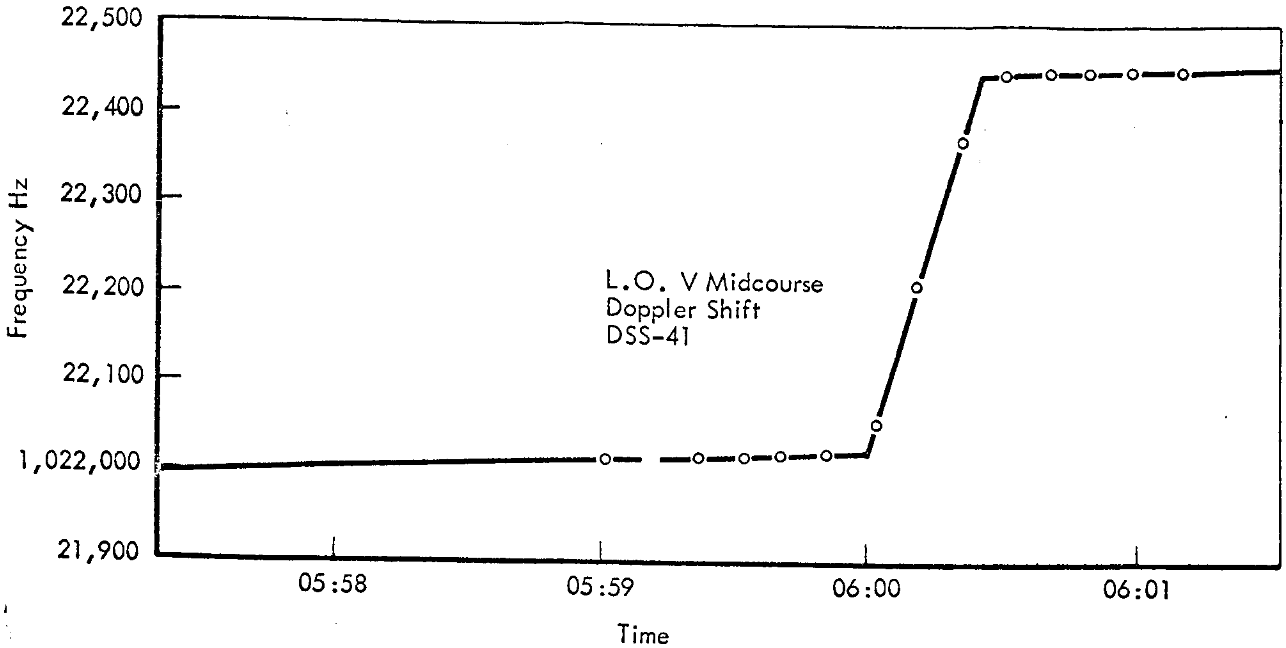


Figure 2-7: Midcourse Doppler Shift

plot. More detailed analysis of the 1-second DSS-12 doppler data indicated that the following engine burn characteristics were actually achieved:

	Commanded	Actual (from Doppler Data)
Ignition time (spacecraft GMT)	215:06:00:00	215:05:59:59.94 ±0.1 sec.
Burn time (sec)	*	26.16 ±0.1
Doppler shift (Hz)	429.5	429.0 ±0.1

* Not applicable because the velocity control subsystem shuts down after application of programmed Δ V.

2.2.3.3 Midcourse Through Deboost

Orbit Determination – Post-midcourse orbit determination procedures used for the first time during Mission IV were used again for this mission, and were instituted to resolve the difficulties associated with the prediction of time of closest approach to the Moon encountered in previous missions. Briefly, two different techniques were used:

Method 1: Solve for state vector only, using doppler and ranging data;

Method 2: Solve for state vector, Earth gravitational constraint, and station locations using an *a priori* covariance matrix and doppler data only.

These two procedures give compatible results and together produce small dispersions in time-of-closest-approach predictions. These dispersions are: Mission II, 40 seconds; Mission III, 30 seconds; Mission IV, 10 seconds; and Mission V, 19 seconds.

Dispersions in other approach parameters, $\overline{B}\cdot\overline{T}$ and $\overline{B}\cdot\overline{R}$, were similar to those experienced for Missions I, II, and III.

Table 2-7 is a chronological sequence of the lunar encounter parameters obtained from the 15 Project orbit determinations performed after midcourse. Final design of the deboost maneuver used OD 2228, which was based on 42 hours of two-way doppler data. A "best estimate" determination done after the deboost maneuver using the last 10 hours of data before deboost indicated that OD 2228 predicted $\overline{B}\cdot\overline{R}$ within 5.0 kilometers, $\overline{B}\cdot\overline{T}$ within 1.0 kilometer, and time of closest approach within 4.5 seconds. As noted above, range unit data were used in the Method 1 fits but not in the Method 2. The Method 2 fits gave ranging residuals on the order of 50 meters. Figure 2-8 shows the aim point dispersions in $\overline{B}\cdot\overline{T}$, $\overline{B}\cdot\overline{R}$, and TCA.

Table 2-7: Lunar Encounter Parameters

Orbit Solution	Solution Type†	$\overline{B}\cdot\overline{T}$ (km)	$\overline{B}\cdot\overline{R}$ (km)	Time of Closest Approach (GMT) Day 217 hr:min:sec	Data Arc Length (hrs:min)
2102	**1**	**380.5**	**5,716.4**	**17:09:34.3**	**3:18**
2104	**2**	**354.4**	**5,718.1**	**17:10:21.2**	**3:18**
2106	**1**	**371.9**	**5,736.0**	**17:10:02.5**	**6:23**
2108	2	354.6	5,719.0	17:10:21.6	6:24
2210	1	356.8	5,699.1	17:10:04.7	12:0
2212	**2**	**351.8**	**5,702.0**	**17:10:15.5**	**12:0**
2214	1	356.6	5,700.2	17:10:05.7	15:30
2216	2	351.4	5,702.8	17:10:16.7	15:30
2318	1	354.5	5,698.8	17:10:08.5	22:18
2320	2	352.2	5,699.7	17:10:13.4	22:18
2322	1	350.9	5,699.6	17:10:15.6	27:17
2324	2	352.8	5,699.9	17:10:12.5	27:17
2126*	1	351.6	5,701.0	17:10:15.1	35:57
2228**	2	350.9	5,701.0	17:10:17.1	42:09
2199***	1	352.2	5,696.0	17:10:12.6	10:00

* Used for deboost maneuver preliminary command conference.

** Used for deboost maneuver final command conference.

*** "Best estimate" determination – used data up to deboost maneuver.

† Solution Type 1 used ranging and two-way doppler data; Solution Type 2 used two-way doppler data with an *a priori* covariance matrix obtained from the premidcourse data.

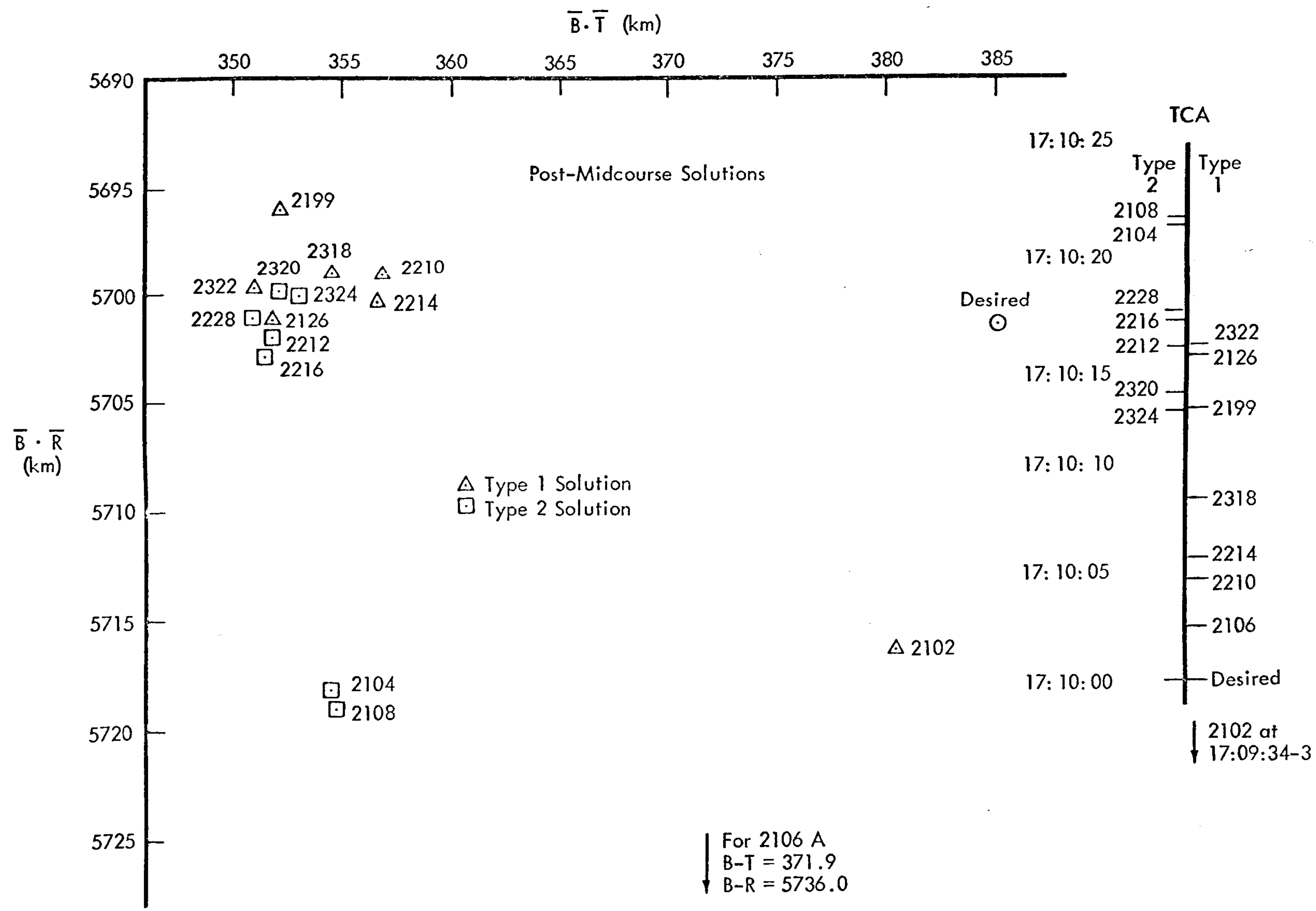


Figure 2-8: Aiming Point Plot

TCA	= 217:17:10:17.121 GMT			
$\overline{B}\cdot\overline{T}$	= 351.0 km	i = 86.5		hyperbola conic elements
$\overline{B}\cdot\overline{R}$	= 5701.0 km	ω = 323.7		
RCA	= 2413.1 km	Ω = 104.68		
V ∞	= 0.940 km/sec			

Deboost Design and Execution — Injection into the initial lunar ellipse was accomplished on August 5, 16:48:54.4 GMT, concluding the cis-lunar phase of Mission V. Final design of the deboost maneuver was achieved with OD 2228 (containing a 42-hour data arc). The above lunar encounter parameters were predicted.

The deboost design had a requirement that the spacecraft satisfy a crosstrack tilt constraint for a given Apollo target site (Site V-8a). Thus, the spacecraft had to be placed over a given longitude and latitude at a particular time in the orbit. The design was made with nominal values for inclination (i), time of illumination, semi-major axis (a), and longitude of ascending node (Ω). The parameters that were permitted to vary to achieve mission requirements were argument of perilune (ω) and perilune radius (r_p). Because of the highly successful midcourse, a near-nominal design of the initial ellipse was possible, as illustrated in the table below.

Execution of the deboost required a two-axis spacecraft attitude maneuver to align the engine thrust vector in the proper direction. Some of the criteria in choosing the particular maneuver (where there are 12 to choose from) were: (1) maintaining maximum sun lock, (2) DSS vector outside of any antenna null regions, (3) minimum total angular rotation, and (4) minimum angular errors. The maneuvers chosen were sunline roll, 20.24 degrees; and pitch, 76.91 degrees.

The engine was ignited at Day 217, 16:48:54.4 GMT after the attitude maneuver had been successfully accomplished. Total burn time

Element	Nominal (Desired)	Design
Epoch	217:17:10:00 GMT	16:52:18.4 GMT
r_a	7,788.1 km	7,788.1 km
r_p	1,938.1 km	1,939.8 km
i	85.0 degrees	85.05 degrees
ω	1.37 degrees	1.35 degrees
Ω	117.76 degrees	117.76 degrees

Epoch	217:16:52:15.562 GMT	
r_a	7,766.4 km	21.7 km low
r_p	1,932.6 km	7.2 km low
ι	85.01 degrees	0.04-degree low
ω	1.55 degrees	0.20-degree high
Ω	117.77 degrees	0.14-degree low

was 498.1 seconds, yielding a ΔV of 643.04 meters per second. As a result of this near-nominal deboost, the orbital parameters of the initial ellipse as computed by OD 4106 (5-hour, 30-minute data arc) are, and differ from the design as shown above.

In the event of engine failure at deboost, a backup fly-by phase was considered. Two distinctly different modes of picture taking were examined. First was a sequence of nine single-frame photos. The first photo would have the camera axis pointed by a three-axis maneuver at the center of the Moon. The attitude of the spacecraft remains fixed while the other eight photos are taken. Later, the camera would similarly be pointed with another three-axis maneuver at the center of the Moon and three sets of four-frame sequences taken at predetermined times.

Doppler Data Monitoring during Deboost Maneuver — Several hours before the deboost maneuver, a set of engine-burn doppler predicts was computed. This computation used the OD 2228 state vector and the nominal maneuver angles and engine performance parameters to predict the doppler shift frequencies before, during, and after the maneuver. These predicted doppler frequencies were then plotted in the region of the burn and the actual doppler shift data plotted on the same curve as the data were received in the SFOF. The resulting plot (Figure 2-9) shows that the maneuver was nominal to the accuracy of the plot. More detailed analysis of the 1-second Station 12 doppler data indicated that the following engine-burn characteristics were actually achieved.

Event	Commanded	Observed (Doppler Estimate)
Ignition time (S/C GMT)	217:16:48:54.4	217:16:48:54.43 $\pm$0.1 sec.
Burn time (sec)	*	498.1 $\pm$0.1
Doppler shift (H z)	1,850.0	1,824.0 $\pm$0.1

* Not applicable because velocity control subsystem shuts down after application of programmed ΔV.

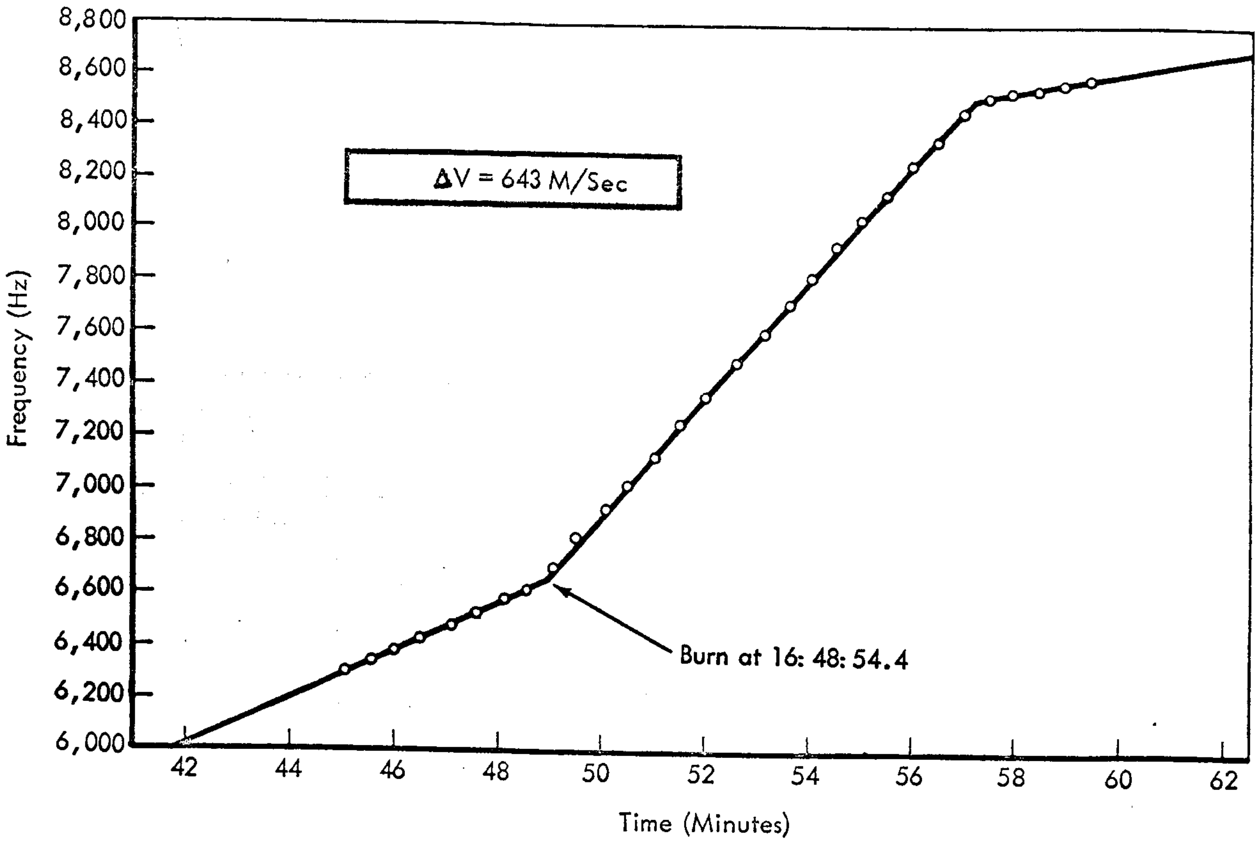


Figure 2-9: Predicted Doppler Shift During Deboost

2.2.3.4 Initial Ellipse Orbit Determination – Comparison of predicted and observed deboost doppler tracking data indicated that the deboost maneuver was near nominal. The first orbit determination after deboost (OD 4102 – 30 minutes of data) and two subsequent determinations confirmed this estimate. The following table shows the desired and obtained orbit elements.

Orbital Element	Deboost Design	First OD Estimate (OD 4102)	Best OD Estimate (OD 4106)
Perilune altitude (km)	201.7	195.3	194.5
Apolune altitude (km)	6,050.0	6,059.7	6,028.3
Inclination (degrees)	85.05	85.03	85.01
Longitude of ascending node (degrees)	117.9	117.8	117.8
Argument of pericenter (degrees)	1.35	1.47	1.55

Orbit determination work during this phase was scheduled to permit command conference flight path control computations to be supported with determinations based on two orbit data arcs (approximately 16 hours of data). Table 2-8 shows which determinations were used for each command conference and the corresponding photo site. The basic procedure used for each determination was:

- Use two orbits of two-way doppler tracking data;
- Use the LRC 7/28B lunar harmonics;
- Solve for the state vector and tailor the basic lunar model by solving for higher order harmonics C32, C33, C42, C43, C44, S32, S33, S42, S43, and S44;
- Place solution epoch at 190 degrees true anomaly.

The LRC 7/28B lunar model was chosen in a premission study from five models as the model that best represented the Moon's gravitational field. All state vector mappings to maneuver times were done using the "tailored" set of harmonics.

No ranging data were obtained during this or subsequent mission phases due to a suspected anomaly in the ranging transponder and/or flight programmer. Preflight procedures had called for use of ranging data only as a check on each solution. The loss of this data did somewhat lower the degree of confidence in the determinations but did not affect solution accuracy. Three-way doppler tracking data were used for checking the solution (it was essentially weighted out of the fit) and were very helpful in detection of erroneous DSS transmitter frequency inputs to the orbit data generation computer program (ODGX).

The executed transfer maneuver computations were based on OD 4118, which used a two-orbit data arc. The transfer execution time was 10 hours, 29 minutes from the last data point. Appendix B, Volume VI of this document series contains detailed summaries of each 10-orbit determination done during this mission phase.

Photography — Four farside photo sites, exposing 18 frames, were photographed in the initial ellipse. All camera pointing maneuvers were two-axis (hroll - pitch) and were chosen such that the camera window was in the shade during the photo sequence. A frame exposure summary for the initial ellipse is given in the following table.

Photo Site	Orbit Number	Frame Numbers
A-1	2	5 to 12
A-2	2	13 to 20
A-3	3	21
A-4	4	22
A-5	4	23 (Blank film set)

High-resolution coverage of these photo sites is shown in Figure 2-11.

Transfer Design and Execution — Another task of the flight path control group during the initial-ellipse phase of Mission V was design of an appropriate first-transfer maneuver. The transfer from initial to intermediate ellipse was executed("stop engine burn")on August 7, 1967, at 08:43:59.2 GMT and resulted in an intermediate ellipse nearly identical to that designed. This event concluded 40 hours in the initial orbit. The following ground rules were used in design of the first transfer.

- Site V-8a to be photographed in Orbit 26 with 5 degrees of crosstrack tilt;
- Final ellipse (immediately after second transfer) to have apolune altitude of 1,500 kilometers and perilune altitude of 100 kilometers;
- Reference target for first transfer is time and location of second transfer;
- Second transfer will occur at the perilune of Orbit 10 and will adjust apolune altitude only (i.e., Hohmann transfer);
- First transfer will occur near the apolune between Orbits 4 and 5 and will adjust the ellipse to meet the above conditions.

Table 2-8: Orbit Determinations Used For Photo Command Conferences

Mission Phase	Photo Site Number	P.C.C.	F.C.C.	Update
Initial ellipse	A-1	(a)	4,106	--
	A-2	(a)	4,106	--
	A-3	(a)	4,106	--
Intermediate ellipse	A-4	4,312	4,314	--
	A-6	(b)	(c)	--
	A-7.1	(c)	(c)	--
	A-8	(c)	(c)	--
	A-9	(c)	(c)	--
	A-10	(c)	(c)	--
	A-11.2	5,214	5,318	--
	A-12	5,214	5,318	--
Final ellipse	A-13	(d)	(e)	--
	A-14	(e)	6,204	--
	V-1	(e)	6,204	--
	V-2.1	6,308	6,312	--
	V-3.1	6,312	6,114	--
	A-15	6,312	6,114	--
	V-4	6,312	6,114	--
	V-5.1	6,114	6,116	--
	V-6	6,116	6,218	--
	A-16.1	6,116	6,218	--
	V-8a	6,116	6,320	6,122*
	V-8b	6,116	6,320	6,122*

Table 2-8 (Continued)

Mission Phase	Photo Site Number	P.C.C.	F.C.C.	Update
Final ellipse	V-9.1	6,218	6,122	6,124*
	A-17.1	6,218	6,122	--
	V-10	6,320	6,124	6,226*
	V-11a	6,320	6,124	6,328
	V-11b	6,122	6,226	6,328
	V-12	6,122	6,226	6,330
	V-13	6,124	6,328	6,330*
	V-14	6,124	6,328	6,330*
	A-18.1	6,124	6,328	--
	V-15.1	6,226	6,330	6,132
	V-16a	6,226	6,330	6,134
	V-16b	6,328	6,132	6,134*
	A-19	6,328	6,132	--
	V-18	6,132	6,236	6,338*
	V-19	6,132	6,236	6,340*
	A-20	6,132	6,340	--
	V-21	6,132	6,340	6,142
	V-22	6,236	6,142	6,144
	V-23.2	6,236	6,142	6,144
	V-24	6,340	6,144	6,246
	V-25	6,340	6,144	6,246
	A-21	6,340	6,144	--
	V-26.1	6,142	6,246	6,348

Table 2-8 (Continued)

Mission Phase	Photo Site Number	P.C.C.	F.C.C.	Update
Final ellipse	V-27a	6,142	6,246	6,348
	V-27b	6,144	6,348	6,350
	V-28	6,144	6,348	6,350
	V-29	6,246	6,350	6,152*
	A-22	6,246	6,350	--
	V-30	6,246	6,350	6,152*
	V-31	6,152	6,254	6,348
	V-32	6,152	6,254	6,348
	V-33	6,356	6,358*	6,350
	V-34	6,356	6,358*	6,350
	V-35	6,358	6,360	6,152
	V-36	6,358	6,360	6,152
	V-37	6,360	6,162	6,256
	A-23	6,360	--	6,256
	V-38	6,358	6,162	6,264
	A-24	6,360	6,264	--
	V-40	6,162	6,266	6,368R*
	V-41	6,162	6,266	6,368R
	V-42a	6,264	6,368R	6,170
	V-42b	6,264	6,368R	6,170
	V-43.2	6,266	6,170	6,172
	A-25	6,266	6,170	--
	V-45.1	6,368R	6,274	6,378
	V-46	6,368R	6,274	6,378

Table 2-8 (Continued)

Mission Phase	Photo Site Number	P.C.C.	F.C.C.	Update
Final ellipse	V-48	6,172	6,378	6,380
	V-49	6,172	6,378	6,182
	V-50	6,378	6,182	6,184
	V-51	6,378	6,182	6,286

(a) Nominal post-deboost state vector.

(b) Nominal post (first) transfer state vector--design based on OD 4312.

(c) Nominal post (first) transfer state vector--design based on OD 4314.

(d) Nominal post (second) transfer state vector--design based on OD 5214.

(e) Nominal post (second) transfer state vector--design based on OD 5122.

* Need for an update was studied, but it was decided not to send it to the spacecraft.

A set of lunar harmonic coefficients generated by Boeing, designated Boeing S-5 harmonics, were used during the transfer design. The final maneuver design was based on a state vector from Orbit Determination 4118.

As a precaution, a backup maneuver was also designed. This maneuver was to be executed only in the event that the prime-transfer maneuver could not be performed. The backup maneuver would have been executed one orbit revolution (about 8.3 hours) later than the prime transfer.

The transfer maneuver was designed by targeting to the three parameters, (1) perilune radius (R_p), (2) longitude of the ascending node (Ω), and (3) orbit inclination (i). The desired values for these targeting parameters were specified at the fifth perilune following the first-transfer maneuver. This also established the location of the second-transfer point. The desired perilune radius, 1,838.09 kilometers, is specified by mission design. The desired longitude of the ascending node, 71.41 degrees, and orbit inclination, 84.75 degrees, adjusts the orbit plane such that the second-orbit adjustment can be a Hohmann transfer. The transfer true anomaly of 200 degrees was chosen to maintain a nominal period.

The attitude maneuvers required to perform this transfer were sunline roll, −63.10 degrees; pitch, 71.29 degrees.

Selection of this attitude maneuver sequence was based on maintaining sun lock as long as possible and compliance with antenna constraints with a minimum of angular rotation. A ΔV of 15.97 meters per second was required. This was well below the budgeted ΔV of 41 meters per second.

Element	Pretransfer Prediction	Preflight Nominal	
Apolune radius	7,805.05 km	7,788.00 km	
Perilune radius	1,838.09 km	1,838.09 km	
Inclination	84.75 degrees	85.08 degrees	Selenographic of date coordinates
Argument of perilune	0.82 degrees	0.46 degrees	
Longitude of ascending node	71.41 degrees	71.41 degrees	

Predicted conic elements at the time of second transfer maneuver, Orbit 10, are given above with the desired nominal values from premission design.

All elements above are given for August 9, 05:09:30.6 GMT.

The predicted conic elements before and after the impulsive transfer maneuver are given below to indicate the change in each caused by the maneuver. All elements are given for August 7, 08:43:54.15 GMT.

Prior to acceptance of this final design of the transfer maneuver, various alternative sets of search parameters were investigated: R_p; R_p, period, i ; R_p, Ω. In each case, some conic element was allowed to deviate to satisfy the search parameters gave results as satisfactory as the set used in the final design – R_p, Ω , i

Dopper Data Monitoring during First Transfer Maneuver – Several hours before the transfer maneuver, a set of nominal transfer maneuver doppler predicts was computed. This computation used the OD 4118 state vector and the designed nominal maneuver to predict the doppler shift frequencies before, during, and after the maneuver. These predicted doppler frequencies were then plotted in the region of the burn and the actual doppler shift data plotted on the same curve as it was received in the SFOF. The resulting plot, Figure 2-10, shows

Element	Pretransfer	Posttransfer	
R_a(km)	7,763.14	7,803.73	
R_p(km)	1,938.81	1,839.42	
i (degrees)	84.55	84.62	Selenographic of date coordinates
ω (degrees)	1.22	1.38	
Ω (degrees)	95.87	95.90	

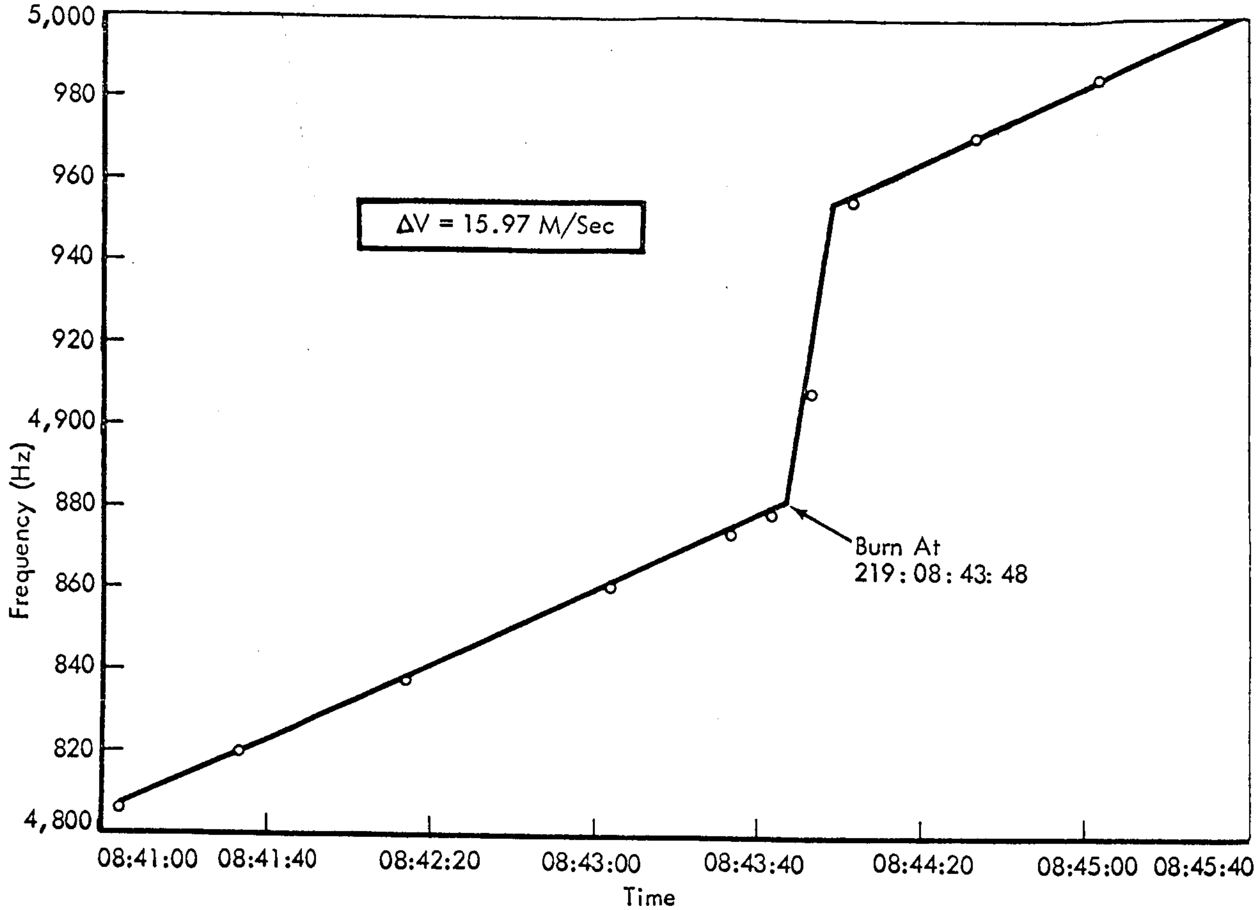


Figure 2-10: Doppler Predicts During Transfer into Second Orbit

the maneuver very near nominal. More detailed analysis of the 1-second DSS-41 two-way doppler data indicated that the following engine-burn characteristics were actually achieved.

	Commanded	Actual (from doppler data)
Ignition time (S/C GMT)	219:08:43:48.7	219:08:43:48.15 $\pm$ 0.01
Burn time (sec)	*	11.4 $\pm$ 0.25
Doppler shift (Hz)	73.0	75.0 $\pm$ 0.1

* Not applicable because velocity control subsystem shuts down after application of programmed ΔV.

	Transfer Design	First OD Estimate (OD 5302)	Best OD Estimate (OD 5214)
Perilune altitude (km)	101.33	118.9	100.4
Apolune altitude (km)	6,065.64	6,075.2	6,066.8
Inclination (degrees)	84.62	84.66	84.61
Argument of perilune (degrees)	1.38	0.903	1.34
Longitude of ascending node (degrees)	95.90	95.91	95.90

2.2.3.5 *Intermediate Ellipse*

Orbit Determination – Comparison of predicted and observed transfer doppler tracking data indicated that the first-transfer maneuver was near nominal. The first orbit determination after transfer to the intermediate ellipse (OD 5302 based on 1 hour, 20 minutes of data) confirmed this estimate. The above table shows the desired and obtained orbital elements after transfer.

The orbit parameters for this second ellipse were similar to those of the first ellipse and the orbit determination procedures were identical. No orbit determination anomalies or unusual events occured during this phase. Twelve orbit determinations were completed; Table 2-8 shows which orbit determination was used for each command conference and the corresponding photo site.

The final transfer maneuver computations were based on OD 5122, which used a 2.25-orbit data arc. The transfer execution time was 12 hours, 15 minutes from the last data point.

Appendix B, Volume VI of this document contains the detailed summaries of each of the 12 orbit determinations done during this phase.

Photography – Photographic activity in the intermediate ellipse consisted of six farside photos and one Earth photo, all single-frame exposures. Orientation of the spacecraft for the first photo taken in the intermediate ellipse (Site A-6) was based on a nominal-transfer maneuver. Spacecraft orientation for all others was based on intermediate-ellipse orbit determinations. As in the initial ellipse, the camera was pointed by using two-axis (hroll-pitch) maneuvers. The camera window for all farside photographs was in the shade during the photo sequence; for the Earth photo, the camera window was on the shaded side of the spacecraft. The following table summarizes the frame exposures in the intermediate ellipse.

Photo Site	Orbit Number	Frame Number
A-6	5	24
A-7	6	25
A-8	7	26
A-9 (Earth)	8	27
A-10	8	28
A-11.2	9	29
A-12	10	30

Farside coverage is shown in Figure 2-11.

Earth Photo – An Earth photo was not originally planned for Mission V. There was, however, a blank film-set frame scheduled near the

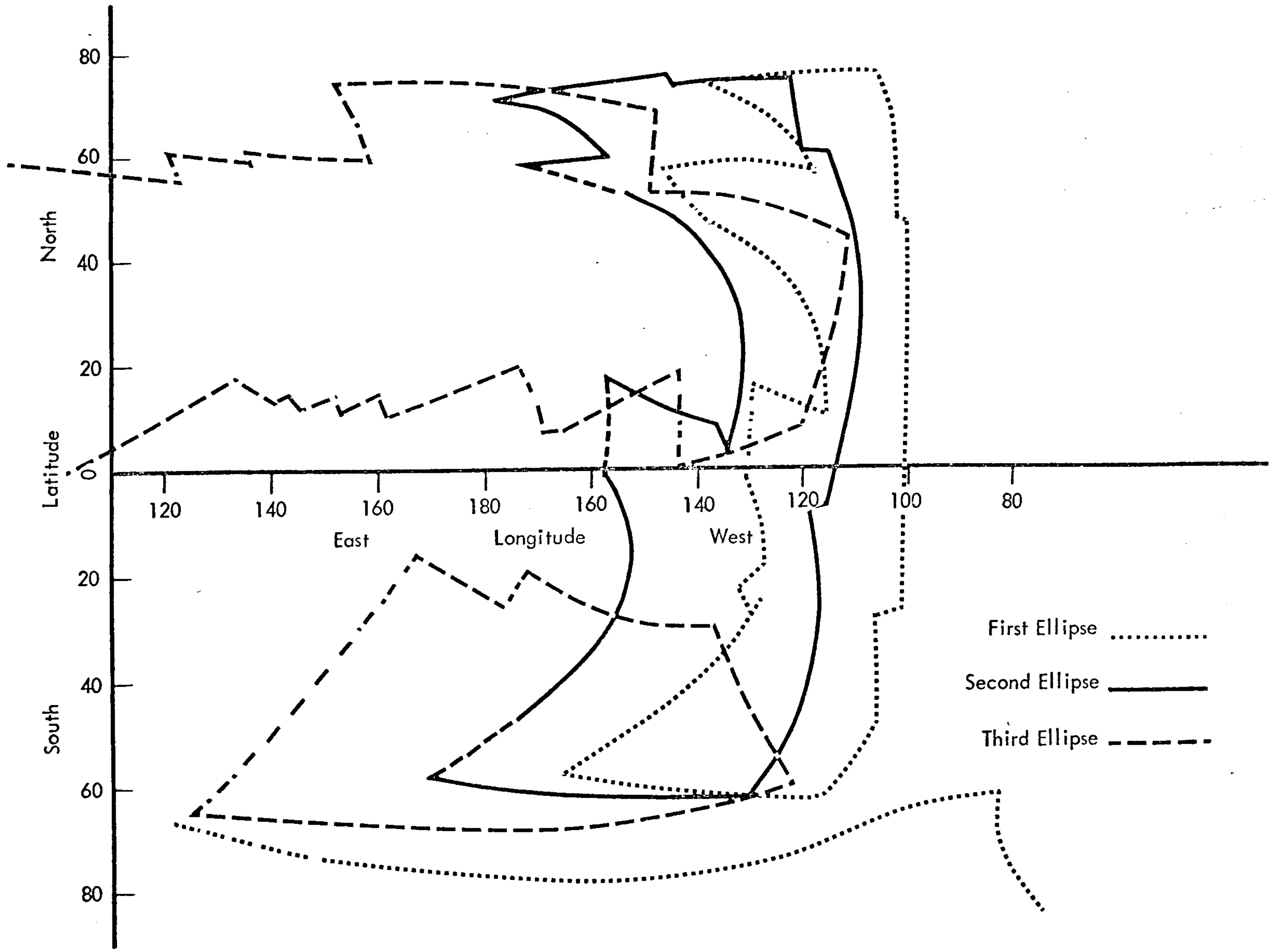


Figure 2-11: Boundary of Areas Photographed on Mission V

apolune between Orbits 7 and 8. At that time, the Earth, as seen from the Moon, would be nearly fully illuminated. It was decided, therefore, to use this otherwise blank frame for an Earth photo. Mission and operations directives were issued on Day 218 (August 6, 1967) implementing this plan. The directives called for a two-axis maneuver to point the camera axis at the center of the Earth, with time of photography no later than 8 hours after the previous photo, since film movement requirements were still to be met. The Earth photo was identified as Photo A-9. Although not specifically stated, it was implied that this photo would not include any of the Moon as in Mission I. First, due to orbit geometry, the Earth never "set" (occulted), making it difficult to include both the Earth and Moon in the same photo. Second, it was designed to expose the film for the Earth alone to obtain the best resolution.

The Earth photo design resulted in the following commands.

- Photo time, Day 220, 09:05:00 GMT;
- Roll, -166.62 degrees;
- Pitch, -38.49 degrees.

This photo time was 7 hours, 23 minutes after the previous photo (A-8), satisfying the film movement constraint. The relative positions of the Earth, Sun, Moon and spacecraft at time of photography are indicated in Figure 2-12, which also shows a view of the Earth indicating the camera-axis intercept, subsolar point, terminator longitude, and edge-of-Earth longitude.

At the time the photo was taken, the spacecraft was 5,870.8 kilometers above the Moon and 10.2 degrees past the apolune beginning Orbit 8. The Earth was 361,730 kilometers away and subtended an angle of 1.985 degrees. The southern tip of India was at the subspacecraft point.

Second Transfer Design and Execution — Transfer from the intermediate to final ellipse took place on August 9, 1967, at 05:11:05.6 GMT. This event concluded nearly 2 days in the intermediate ellipse.

The ground rules followed for design of the second transfer maneuver were to photograph Site V-8a from the perilune of Orbit 26 with 5 degrees of crosstrack tilt, and to have apolune altitude of 1,500 kilometers and perilune altitude of 100 kilometers for final ellipse (immediately after second transfer).

The Boeing S-5 lunar harmonic coefficients were used in the second-transfer design. The final design was based on the state vector from Orbit Determination 5122.

Because of the execution accuracy of the first-transfer maneuver, a Hohmann second transfer satisfied all ground rules. The maneuver was designed by targeting to a semi-major axis of 2,537.09 kilometers at the next apolune and constraining engine burn to occur at perilune. This maneuver would reduce apolune altitude from 6,080 to 1,499 kilometers, assuming an impulsive burn. However, due to the long engine burn time required to perform this maneuver, finite burn effects modified the resulting apolune altitude to 1,500 kilometers as required for final photo design. The attitude maneuvers required were sunline roll, −84.02 degrees; and pitch, −96.79 degrees. Maneuver selection was based on the same criteria used for the first transfer.

The orbit change required a ΔV of 233.67 m/sec out of a budgeted 292 m/sec and took 152.9 seconds to complete.

Predicted conic elements before and after the burn follow, indicating the change in each caused by the maneuver.

Elements	Pretransfer	Posttransfer
Epoch, Aug. 9, 1967 GMT	05:08:32.65	05:11:03.81
R_a (km)	7,821.07	3,239.97
R_p (km)	1,837.22	1,837.20
i (degrees)	84.75	84.76
ω (degrees)	0.90	1.61
Ω (degrees)	71.40	71.40

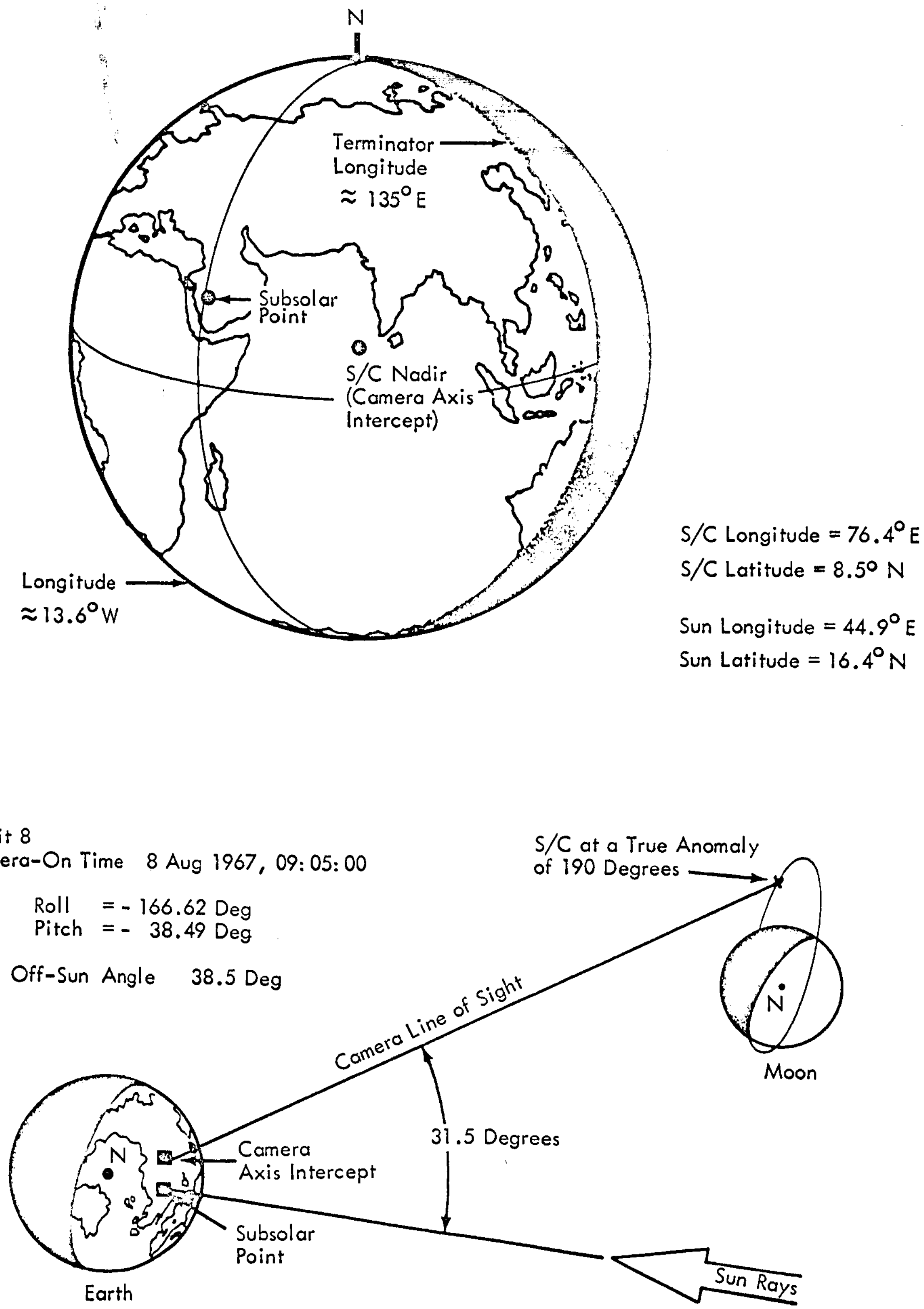


Figure 2-12: Site A-9 Earth Photo Geometry

Element	Pretransfer Prediction	Preflight Nominal	Actual
Epoch (GMT)	223:08:10:13.87	223:08:15:21.0	223:08:08:12.8
Apolune radius (km)	3,239.41	3,238.5	3,239.7
Perilune radius (km)	1,836.87	1,837.6	1,835.2
Orbit inclination (degrees)	85.13	85.11	85.00
Longitude of ascending node (degrees)	43.28	44.19	43.27
Argument of perilune (degrees)	1.01	0.07	1.23
Spacecraft longitude (degrees)	43.36	43.32	43.37
Spacecraft latitude (degrees)	1.01	0.06	1.23
Crosstrack tilt to longitude of Site V-8a	4.62	5.00	4.47

A comparison of predicted and nominal orbital elements and spacecraft position at the perilune of Orbit 26 is given above.

The comparison gives an indication of the accuracy of design of this and execution of all previous velocity maneuvers.

A backup second-transfer maneuver was also designed in the event the prime maneuver could not be performed. This backup maneuver would have been executed one orbit revolution (about 8.3 hours) later than the prime transfer.

Doppler Data Monitoring during Second Transfer—Several hours before the transfer maneuver, a set of nominal transfer maneuver doppler predicts was computed using the OD 5122 state vector and the designed nominal maneuver to predict the doppler shift frequencies before, during, and after the maneuver. These predicted doppler frequencies were then plotted in the region of the burn and the actual doppler shift data plotted on the same curve as it was received in the SFOF. The resulting plot (Figure 2-13) shows that the maneuver was near nominal. More detailed analysis of the 1-second DSS-41 two-way doppler tracking data indicated that the engine burn characteristics shown below were actually achieved.

2.2.3.6 *Final-Ellipse Phase*

The final-ellipse phase extended from end of second-transfer burn through termination of final readout.

	Commanded	Actual (from Doppler Data)
Ignition time (S/C GMT)	221:05:08:32.7	221:05:08:32.58 ± 0.1
Burn time (sec)	*	152.9 ± 0.1
Doppler shift (Hz)	500.0	524.5 ± 0.1

*Not applicable; VCS shuts down after application of programmed ΔV.

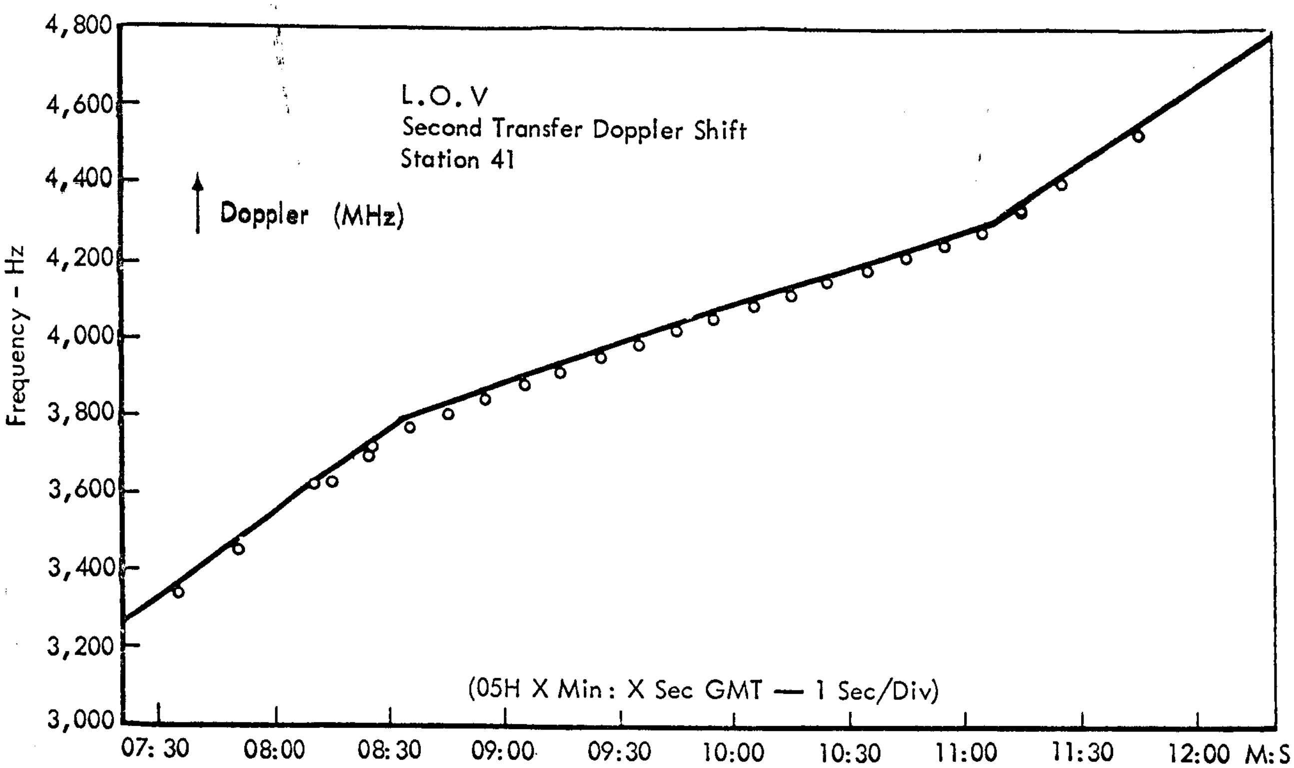


Figure 2-13: Second Transfer Doppler Shift – Station 41

The principal FPAC tasks in this phase included a high-quality orbit determination prior to each primary photo event, design of camera pointing maneuvers and camera-on times, and trajectory predictions, including station rise and set time.

The basic schedule of FPAC tasks, broken down among the three general groups above, is shown in Figure 2-14.

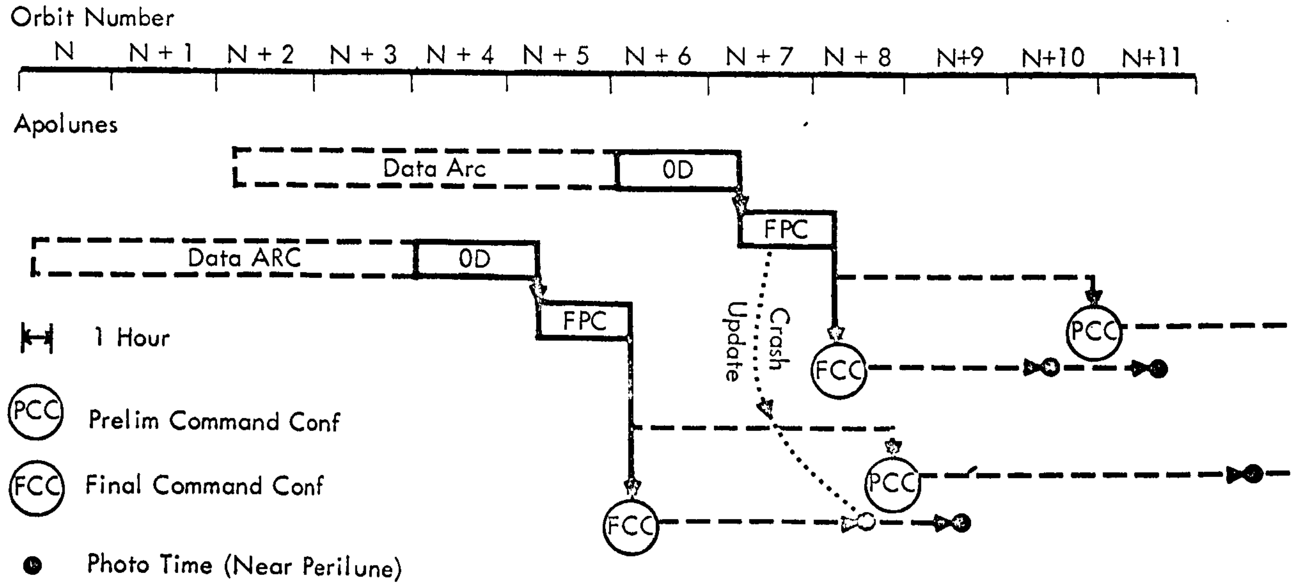


Figure 2-14: Schedule of FPAC Tasks

Orbit Determination — Comparison of predicted and observed transfer doppler tracking data indicated that the transfer was near nominal. The first orbit determination after transfer (OD 6202 based on 1 hour, 16 minutes of data) confirmed this estimate. The following table shows the desired and obtained orbit elements after transfer.

Orbit determination work during this phase was scheduled so that command conference flight path control computations could be supported with determinations based on four orbit data arcs (13 hours of data). Table 2-8 shows which determinations were used for each command conference and the corresponding photo site. Figures 2-15 through 2-18 show the history of the orbital elements, as determined by ODPL from deboost to end of readout.

The basic procedure used for each determination was:

- Use four orbits of two-way doppler tracking data;
- Use the LRC 7/28B lunar harmonics as the basic lunar model;
- Solve for state vector and "tailor" the basic lunar model by solving for higher-order harmonics C32, C33, C42, C43, C44, S32, S33, S42, S43, and S44;
- Place solution epoch 40 minutes after apolune (220-degree true anomaly).

This is essentially the same procedure used in the first two phases, with the exception of data arc length. All state vector mappings were done with the "tailored" harmonics.

Forty-four determinations were done during the photo-taking part of this mission phase. Detailed summaries of each of these determinations may be found in Appendix B, Volume VI of this document series.

During the final orbit phase, "crash" updates were frequently used to redefine the camera-on time specified at the final command conference. These updates were supported by a last possible OD solution (Table 2-8); scheduling of the crash updates is indicated in Figure 2-14.

Photo Design — Nearside and farside photographic coverage was obtained during the final-ellipse portion of Mission V. Forty-one nearside sites were covered in 50 different passes, with five cases of multiple-pass coverage of the same site (two-, three-, and four-pass combinations). The nearside sites were identified by a V-prefix. In addition, 12 farside pictures were taken from near apolune of the final orbit. These sites were identified by an A-prefix.

Orbital Elements	Transfer Design	First OD Estimate (OD 6202)	Best OD Estimate
Perilune altitude (km)	99.1	98.51	98.93
Apolune altitude (km)	1,501.9	1,499.0	1,499.4
Inclination (degrees)	84.76	84.74	84.76
Longitude of ascending node (degrees)	71.40	71.39	71.38
Argument of pericenter (degrees)	1.61	1.80	1.88

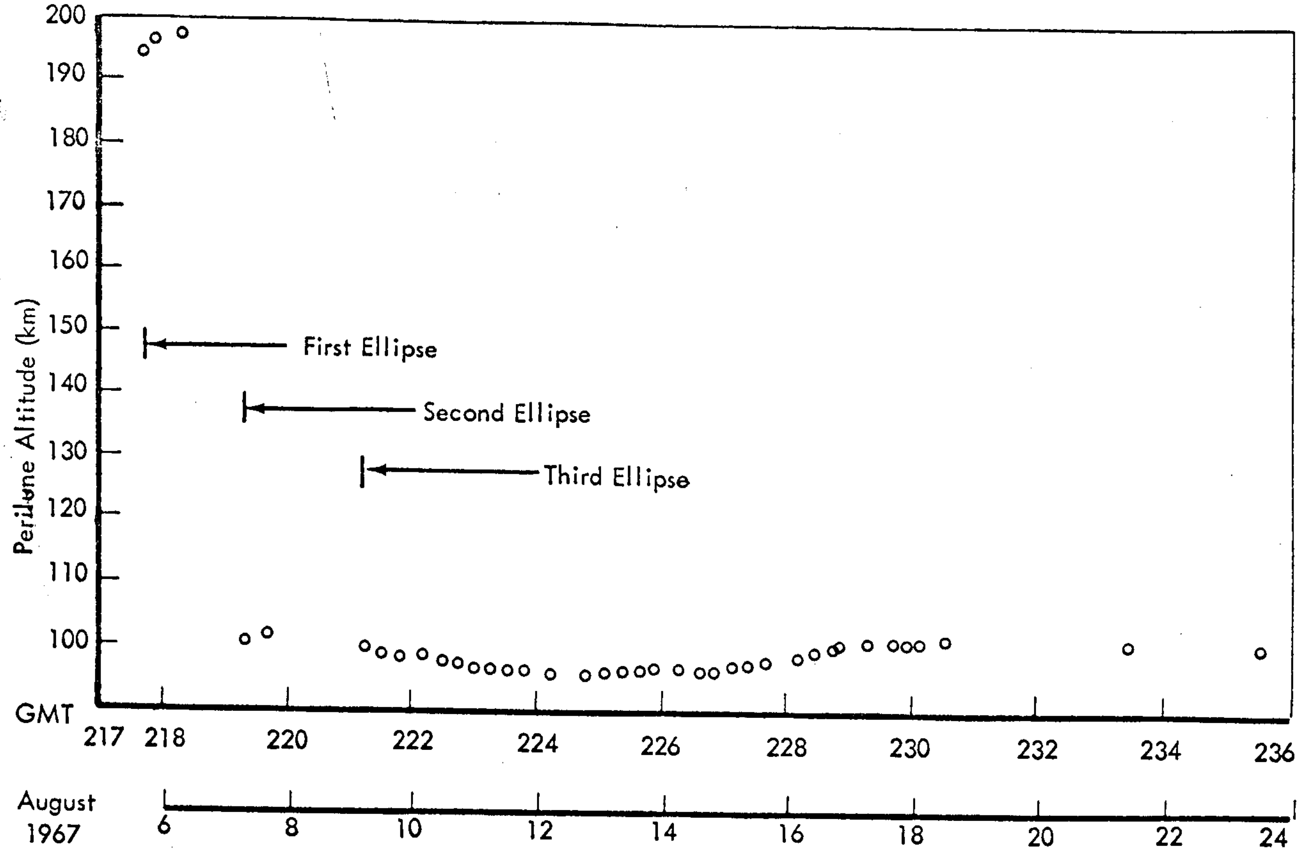


Figure 2-15: Perilune Altitude vs Time

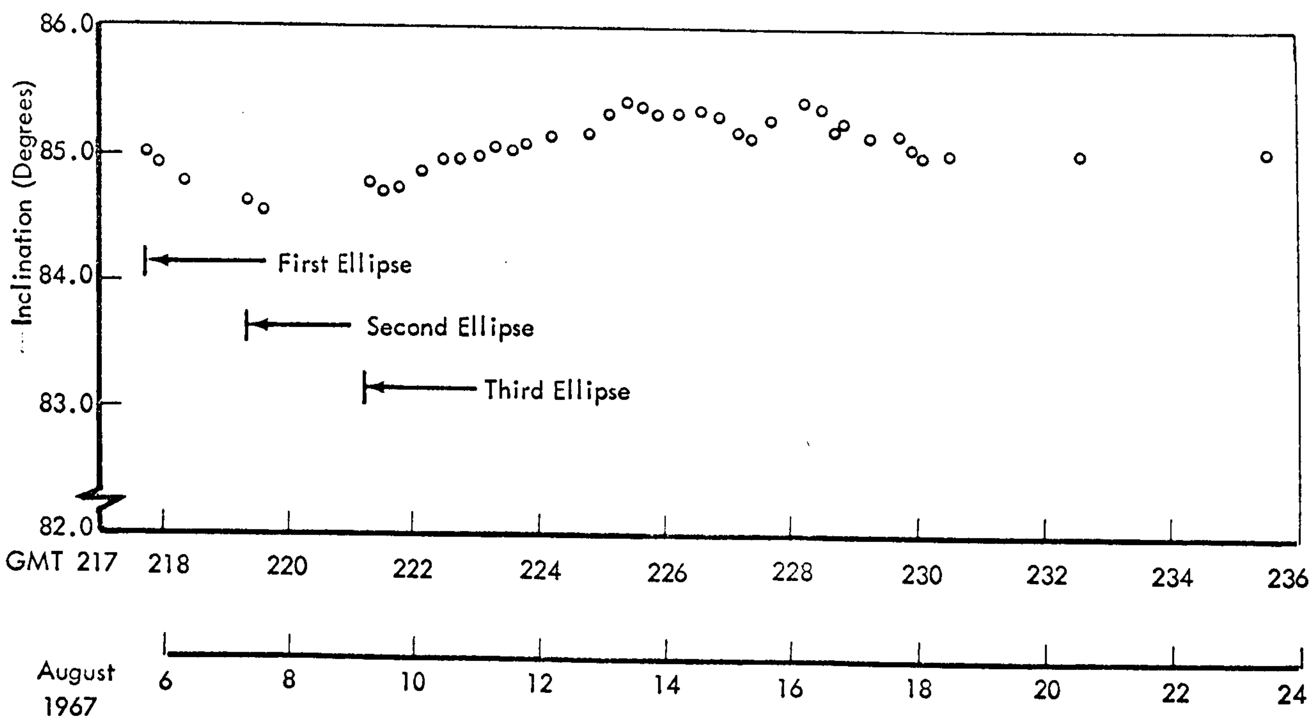


Figure 2-16: Ellipse Inclination vs Time

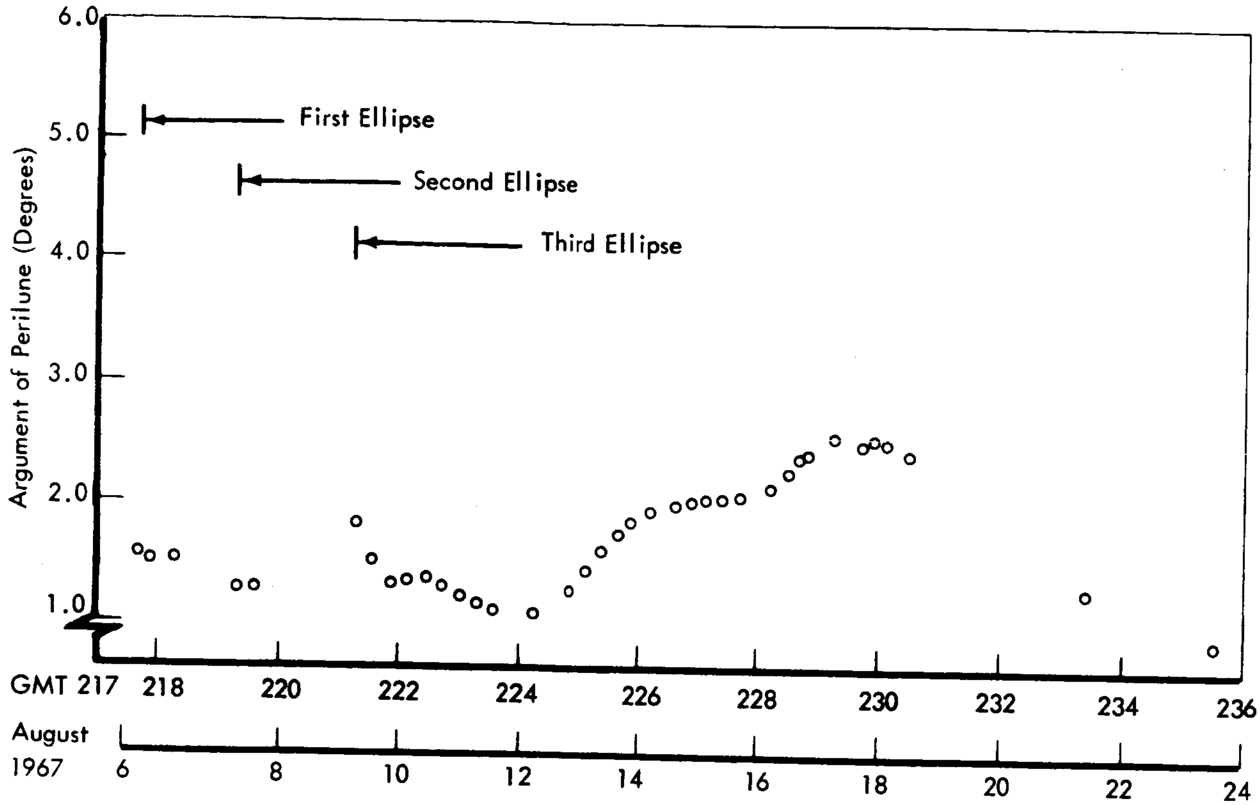


Figure 2-17: Argument of Perilune History

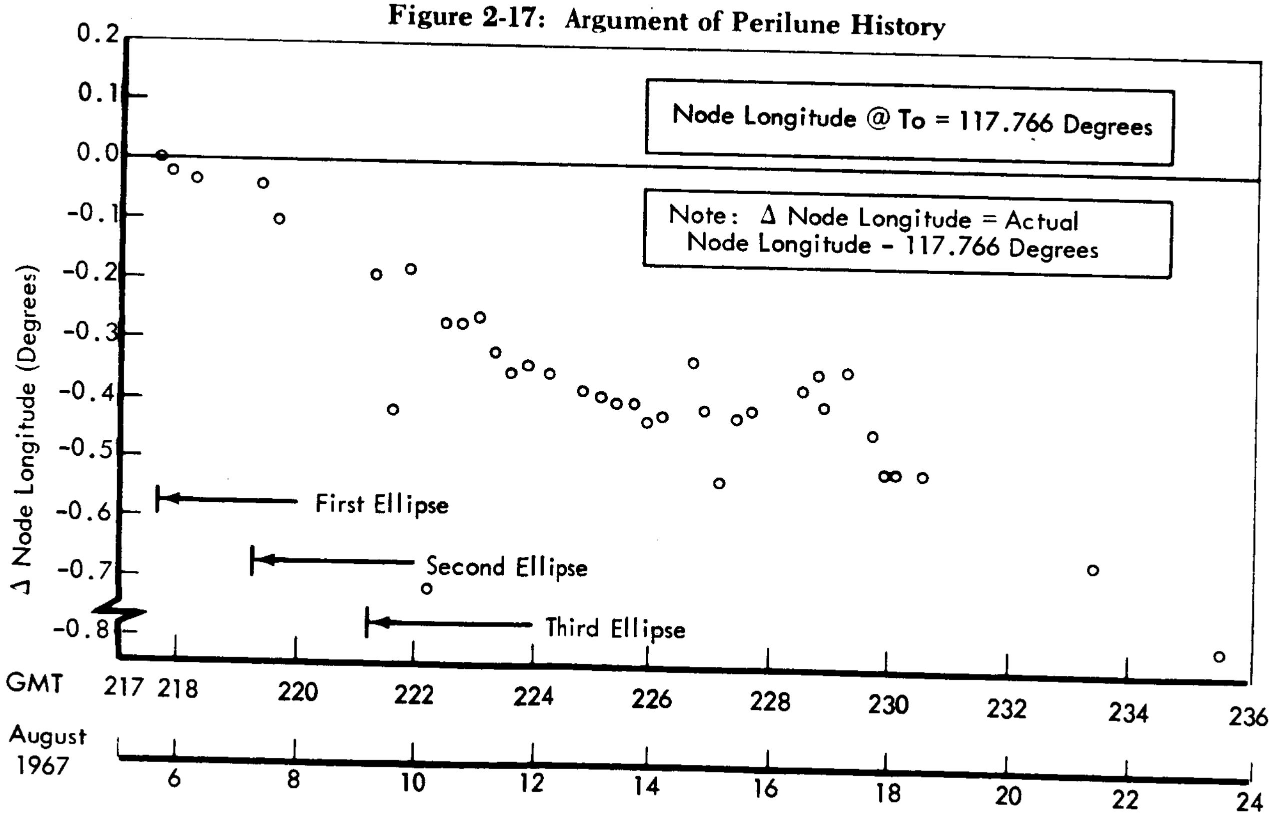


Figure 2-18: Inertial Node Longitude History

A total of 174 photo frames was exposed for nearside coverage and 12 frames for farside coverage. The site number/orbit number correspondence for final-ellipse photography is given in Table 2-9, which also lists coordinates of each site photographed, the priority, number of frames, and type of coverage. This is the basic specification FPAC used for the photo design.

With one exception, design of the attitude rotations was based on the following ground rule: Aim camera axis so that it points directly at the specified target at the midpoint of the series of exposures; provide for image motion compensation by causing the short axis of the picture frame to be in the same direction as the image motion. (The exception was Site V-25, the Alpine Valley picture. In this case, the camera axis was pointed as specified above, but the picture frame orientation was designed so the long axis of the frame was parallel to the valley.)

Camera aiming types were classified in the following manner.

- Near-vertical – Site was within one orbit spacing of the spacecraft nadir.
- Conventional oblique – Site was more than one orbit spacing remote from the spacecraft nadir.
- Westerly oblique – Site was three to seven orbit spacings remote from the spacecraft nadir.
- Convergent telephoto stereo (same definition as near-vertical or conventional oblique, with the qualification that coverage was of the same area as a previous Mission V photo).

Photo timing was constrained in the following ways.

- For all except V-25 and the westerly oblique sites, the spacecraft at photo time was at the point of closest ground trace approach to target.
- For V-25, the photo was taken at the time when the spacecraft was directly over an extension of the valley centerline.
- For the Westerly obliques, the photo was taken when the spacecraft was at the same latitude as the point target.

The farside photos, which had A-prefixes, were roll-pitch maneuvers designed to aim the camera at a specified point on the Moon, generally west of the spacecraft nadir, at a time when the latitude was a specified value. Medium- and high-resolution coverage of the farside photos taken in the final ellipse is shown in Figure 2-11.

Resultant attitude rotations and camera-on times recommended by FPAC are listed in Appendix B, Volume VI of this document.

Lunar harmonics influenced photo maneuver design by affecting the mapping forward from OD epoch to several minutes before camera-on time. The harmonics used were the set solved for in the particular OD solution: LRC 7/28B, tailored. The LRC 11/11 set was used during the several minutes before camera-on time.

Table 2-9: Final-Ellipse Photography Site Data

Site Number	Orbit Number	Latitude	Longitude	Name	Coverage
A-13	11	28° 31' N	135° 46' W	Farside	1
A-14	13	24° 28' N	137° 38' W	Farside	1
V-1	15	25° 10' S	60° 40' E	Petavius	F4
V-2.1	17	19° 25' S	57° 05' E	Petavius B	1
V-3.1	19	1° 00' S	42° 56' E	IP-1	1
A-15	19	39° 02'	159° 26' W	Farside	1
V-4	20	31° 50' S	51° 50' E	Stevinus A	1
V-5.1	21	2° 10' S	47° 16' E	Messier	1
V-6	23	1° 00' S	42° 56' E	IP-1	1
A-16.1	25	47° 44' S	151° 14' W	Farside	1
V-8a	26	1° 00' S	42° 56' E	IP-1	F4
V-8b	27				F4
V-9.1	28	2.67° N	35.76° E	IIP-2	1
A-17.1	29	48° 59' N	175° 48' W	Farside	1
V-10	30	28° 00' S	27° 45' E	Altai Scarp	1
V-11b	31	2° 40' N	34° 14' E	IIP-2	F4
V-11a	32				F4
V-12	33	0° 26' S	32° 43' E	Censorinus	1
V-13	34	0° 45' N	23° 51' E	IIP-6	1
A-18.1	35	47° 23' S	170° 10' W	Farside	1
V-14	35	22° 12' N	29° 20' E	Littrow	F4
V-15.1	36	17° 12' N	26° 20' E	Dawes	1
V-16a	37	0° 45' N	23° 51' E	IIP-6	F4
V-16b	38				F4
A-19	39	38° 50' N	168° 04' E	Farside	1

Table 2-9 (Continued)

Site Number	Orbit Number	Latitude	Longitude	Name	Coverage
V-18	41	2° 42' N	18° 00' E	Dionysius	F4
V-19	42	14° 50' S	14° 00' E	Abulfeda	1
A-20	44	38° 43' N	150° 50' E	Farside	1
V-21	45	38° 30' N	13° 30' E	South of Alexander	F4
V-22	46	21° 00' N	9° 20' E	Sulpicius Gallas rilles	F4
V-23.2	47	8° 03' N	6° 0' E	Hyginus rilles	F4
V-24	48	4° 45' S	4° 05' E	Hipparchus	F4
V-25	49	48° 55' N	3° 00' E	Alpine Valley	1
A-21	49	38° 35' N	151° 45' E	Farside	1
V-26.1	50	25° 52' N	3° 00' E	Hadley rille	
V-27a	51	0° 25' N	1° 06' W	IIP-8	F4
V-27b	52				F4
V-28	53	13° 40' S	4° 10' W	Alphonsus	F4
V-29	54	12° 50' N	4° 00' W	Rima Bode II	F4
A-22	54	38° 27' N	143° 37' E	Farside	1
V-30	55	41° 45' S	11° 30' W	Tycho	F4
V-31	56	49° 30' N	2° 40' W	Sinuous rille east of Plato	S4
V-32	57	13° 25' N	10° 35' W	Eratosthenes	S4
V-33	59	6° 25' N	14° 45' W	Area of Copernicus CD	1
V-34	60	7° 12' S	16° 45' W	Crater FRA Mauro	F4
V-35	61	14° 40' N	16° 15' W	Copernicus Secondaries	S4

Table 2-9 (Continued)

Site Number	Orbit Number	Latitude	Longitude	Name	Coverage
V-36	62	6° 52' N	18° 15' W	Copernicus H	F4
V-37	63	10° 25' N	20° 18' W	Copernicus	F8
A-23	64	38° 11' N	127° 04' E	Farside	1
V-38	65	32° 40' N	22° 00' W	Imbrium flows	F4
A-24	67	38° 08' N	122° 12' E	Farside	1
V-40	69	13° 10' N	30° 55' W	Tobias Mayer dome	F4
V-41	70	30° 25' S	37° 25' W	Vitello	1
V-42a	71	3° 30' S	36° 11' W	IIIP-11	F4
V-42b	72				F4
V-43.2	73	16° 52' S	40° 00' W	Gassendi	S4
A-25	74	41° 51' N	109° 59' E	Farside	1
V-45.1	76	35° 55' N	41° 30' W	Jura domes	F4
V-46	77	27° 15' N	43° 38' W	Harbinger Mts	F8
V-48	79	23° 15' N	47° 25' W	Aristarchus	F8
V-49	80	25° 09' N	49° 30' W	Cobra Head	F4
V-50	82	28° 00' N	52° 45' W	Aristarchus Plateau	F4
V-51	83	13° 45' N	56° 00' W	Marius Hills	F8

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3.0 Spacecraft Performance

Performances of the individual subsystems aboard Lunar Orbiter V are summarized below. A brief description of each subsystem is also presented. For more detailed configuration and functional information on each subsystem, NASA Document CR 66342 *Lunar Orbiter I Final Report — Mission System Performance,* should be consulted.

The key events of the primary mission are tabulated in Table 2-3.

Launch through Cislunar Injection — Launch vehicle liftoff occurred at 22:33:00.338 GMT on Day 213 (August 1, 1967). First- and second-stage booster performance was as programmed and Lunar Orbiter V was injected into cislunar trajectory at the end of the Agena second burn at approximately 33 minutes after liftoff. Separation from the Agena followed approximately 3 minutes later.

Cislunar Injection through Lunar Injection — DSS-51 (Johannesburg) acquired the vehicle in one-way lock 34.1 minutes after liftoff. DSS-41 (Woomera) acquired the spacecraft in one-way lock 46 minutes after launch and two-way lock 4 minutes later. The first good data after DSS-41 acquired one-way lock verified antenna and solar panel deployment. The first track of Canopus for roll reference was obtained approximately 7 hours after liftoff. The propellant line bleed and propellant squib valve firing events were conducted successfully at approximately 16 hours after liftoff to prepare the velocity control system for the midcourse correction maneuver.

The midcourse maneuver was successfully accomplished 31 hours, 27 minutes after launch with a velocity change of 29.75 meters per second.

Spacecraft injection into lunar orbit was performed at 90 hours, 16 minutes after liftoff. The velocity control rocket engine operated for 498.1 seconds, producing a velocity change of 643.04 meters per second, as programmed. The spacecraft was put into an orbit with an apolune altitude of 6,028.27 kilometers, a perilune altitude of 194.55 kilometers and an inclination of 85.01 degrees to the lunar equator.

Initial-Ellipse Orbits — To verify photo subsystem operation, the Goldstone film was read out approximately 1 hour, 30 minutes after lunar injection. The initial film advance and first photographs were taken in Orbit 2.

Transfer to the intermediate ellipse was performed in Orbit 5. The velocity control rocket engine operated for 10.78 seconds, producing a velocity change of 15.97 meters per second, as programmed.

Intermediate-Ellipse Orbits — Sites A-6 through A-12 were photographed in the intermediate-ellipse orbits. The first priority readout occurred in Orbit 6.

Transfer to final-ellipse orbits was performed in Orbit 10 with a velocity change of 233.66 meters per second, and a velocity control rocket engine operating time of 152.9 seconds.

Final-Ellipse Photography — To keep spacecraft temperatures at nominal values and decrease the rate of thermal paint degradation, the spacecraft was pitched off sunline from 38 to 41 degrees during most of the final-ellipse photography. The normal procedure was to acquire the Sun from the thermal pitch off attitude just prior to the lunar South Pole. The roll attitude was then measured by cycling the Canopus tracker on and off, and the roll maneuver for photography was adjusted by the roll error plus the expected gyro drift between the measurement time and the time for the roll maneuver, which was always the first photographic maneuver. Following photography, the final pitch maneuver was programmed to place the spacecraft at the desired thermal pitch-off attitude.

All spacecraft systems performed nominally from liftoff through final-ellipse photography, and all frames through 217 were exposed per the mission plan.

Final Readout – The Bimat-cut sequence was initiated in a normal manner by starting processing and loading a "Bimat cut" command in the programmer to be executed 34 minutes later. "Bimat clear" was indicated 5 minutes before Bimat-cut was executed, due to Bimat exhaustion.

Very little data were lost as a result of the short Bimat since only one wide-angle frame out of an eight-frame sequence was lost (7.3% of the ground coverage of the last site) due to not being processed.

Final readout was started in Orbit 85 with Frame 217 and proceeded normally through all photographic data to Frame 5.

After readout, rewind of the film was continued to place the splice on supply spool and cover it with two wraps of leader to take it out of tension. Before rewind was completed, the leader apparently parted in or near the readout gate. This failure eliminated the possibility of re-reading out the film but had no effect on the success of the mission.

3.1 PHOTO SUBSYSTEM PERFORMANCE

3.1.1 Description

The photo subsystem is designed to photograph the lunar surface, process the exposed film, scan the processed film with a flying-spot scanner, and provide video signals to the communications subsystem for transmission to Earth. A detailed description of the photo subsystem is contained in NASA Document CR-847, *Lunar Orbiter I Final Report-Photography.* The photo subsystem parameters used for Mission V are:

610-mm lens and folding mirror transmission – 66%
80-mm lens transmission – 91%
80-mm filter density = 0.18
610-mm shutter speeds.
1/25 exposure time –0.036 sec.
1/50 exposure time –0.019 sec.
1/100 exposure time –0.010 sec.
80-mm shutter speeds: Assumed nominal

These parameters were used with the albedos as discussed in the section on exposure control contained in Volume II of this document series.

3.1.2 Subsystem Performance

The photo subsystem performed in an entirely satisfactory manner, in terms of mission objectives. With two exceptions, problems encountered during the photographic and final readout phases of the mission were minor in nature and caused only minimal data loss. Although the apparent short Bimat condition and the leader break during rewind were potentially very serious, they had almost no effect on the photo mission. These and other aspects of photo subsystem performance are discussed below.

Thermal History – Photographic subsystem (PS) temperatures during the prelaunch countdown were kept 10°F higher than previous missions to allow for the anticipated drop in all spacecraft temperatures, after liftoff, due to the additional spacecraft mirrors. Temperatures at liftoff were between 55 and 58°F. The minimum temperatures after liftoff occurred 9 hours later when the 610-mm-lens window temperature was 43.9°F and the Thermal Fin Temperature (PT06) was 46.9°F. Then the "night heaters" were enabled. The heaters were inhibited at Day 214, 23:12 GMT, released again at Day 215, 04:10 GMT, and were left enabled for the rest of the mission (except during photography). During cislunar cruise, the temperatures followed a slightly higher profile than prior missions, ranging around 58°F. After photography started, the PS temperatures began a gradual rise from 68 to 76°F at the time of Bimat cut. At the end of final readout, the temperature was 81°F. Temperature fluctuations were smaller during Mission V than during previous missions.

Camera Thermal Door Test – Because of the thermal door problems encountered on Mission IV, test of the camera thermal door was conducted during cislunar cruise starting at Day 215, 20:15 GMT, to test the door's operation. The door was opened three times, for 1 minute each time, using the normal door operation sequence. Window temperature, "camera door open" discrete, and "camera door close" discrete telemetry channels were monitored for proper operation. The camera thermal door operated normally for this test and all subsequent photo passes, for a total of 77 actuations.

Bimat Clear – The Bimat-cut sequence was initiated in a normal manner. Processing was started and, after 29 minutes, a Bimat-clear signal appeared on the telemetry. This was 5 minutes before the Bimat cut was scheduled to occur. It was concluded that the Bimat supply was less than expected and that the end of the Bimat had passed through the processor. This fact was confirmed during readout when a pattern caused by the knurled Bimat supply core was observed. Based on prelaunch data, it was expected that there were 4 or 5 feet more Bimat, which would have allowed several wraps on the supply spool at Bimat cut. Very little data was lost as a result of the short Bimat. One wide-angle frame was not processed and another was degraded by the knurl marks on the Bimat. These two wide-angle frames were part of an eight-frame sequence. Neglecting the knurl marks, 6.8% of the ground coverage of the last side was missed.

Lost Data – Two framelets of Frame 217 were not recovered during final readout. Readout Sequence 088, the first sequence of final readout, was terminated by a stored-program command, but the readout did not completely terminate. The optical-mechanical scanner stopped in the focus-stop position but scanning and readout electronics did not shut down. In this situation, readout is terminated by a stored-program backup "readout drive on" command. Two framelets are then scanned and readout stops when the optical-mechanical scanner reaches the spot-stop position.

The high-gain antenna was correctly pointed for maximum signal strength for an off-Sun readout. Immediately following the stored-program "readout drive on" command was a stored-program "acquire Sun" command. In this case, after the "acquire Sun" command had been initiated, the high-gain antenna would no longer point at the DSS. Under these conditions, noisy data and even loss of lock could occur if video modulation (Mode 2) was on. Therefore it was necessary to execute "Mode 2 off" prior to the stored-program "readout drive on." Two framelets were scanned after "readout drive on," but the information was not transmitted to the DSS because Mode 2 was off. Prior to the next readout, a decision was made to continue readout without retrieving the two lost frames.

Video Dropout – During Mission V, a phenomenon was observed during readout that had not occurred during previous missions. The problem manifested itself as three to six GRE scan lines with no video on them. Another three to five lines were required for the video to return to normal, often with an associated "S" pattern. No GRE sync loss was evident and everything else appeared normal. This phenomenon was first observed during readout of the Goldstone test films. The first occurrence during active scan of a lunar scene was during Sequence 009. The video dropouts appeared as a thin white line on the GRE films and obscured very little data. Even when several dropouts occurred per framelet, the amount of data lost was not significant, compared with 17,000 scan lines per framelet.

Although several framelets read out late in final readout had over 50 dropouts, these framelets were recorded in priority readout with no dropouts. During final readout there were 52 framelets read out, with 10 or more dropouts per framelet. Of these, 32 were read out in priority readout with no dropouts. All framelets with more than 20 dropouts were previously read out without dropouts. Seven frame pairs had 50 or more dropouts but again six of these were readout in priority readout without dropouts. Frame 84 was not previously read out but had less than 15 dropouts in any one framelet.

Rewind Anomaly – After final readout was completed, the film was to be wound on the supply spool so that only leader would be left in the camera. This was done to avoid film set and enable further readout, if necessary. Before rewind was completed the leader apparently parted in or near the readout gate. This conclusion was reached after the readout looper abruptly went to mechanical full and the video output voltage went to its maximum value, indicating no film in the readout gate. Twenty frames were then advanced into the camera storage looper from the supply reel. The looper did not dump into the readout looper, which it would have done had the readout looper indication been in error. It was concluded that the leader had torn. There was no immediate indication of the cause for the failure. Photo subsystem operations were then terminated.

3.2 COMMUNICATION SUBSYSTEM

The communications subsystem performed satisfactorily throughout the mission. All video and telemetry data were successfully transmitted during the mission.

The gain margins for both the high- and low-gain modes of operation were in accord with the link analysis (Boeing Specification 10-70060) and varied as predicted with antenna pointing and spacecraft attitude.

The transponder was operated in an "off-set frequency" mode for most of the mission to avoid possible interference with Lunar Orbiters II and III, both of which were still being tracked.

The TWTA was left on during most of the mission. It was turned off only when operation might have resulted in damage to the tube (vibration or low bus voltage). This occurred when the spacecraft was maneuvered for velocity change sequences. At the conclusion of the prime mission, the TWTA had been turned on and off six times with a total operating time of 469.63 hours.

3.2.1 Launch through Injection (Cislunar)

Launch vehicle liftoff occurred at Day 213, 22:33:00.338 GMT with an azimuth heading of 104.8 degrees and the subsystem performing normally. Telemetry data received via the Agena interface provided real-time data at the SFOF from liftoff to 35.56 minutes after launch. Cislunar injection occurred 32.79 minutes after launch and the communications subsystem was functioning normally in Modulation Mode 3.

3.2.2 Cislunar Injection to Injection (Lunar)

Cislunar injection occurred 32.79 minutes after launch and 1.21 minutes prior to the first S-band acquisition by DSS-51 (Johannesburg). Acquisition reports show that DSS-41 acquired the spacecraft one-way 46 minutes after launch at a signal strength of −131 dbm on the SAA (S-band acquisition aid) antenna. This acquisition occurred 11.02 minutes after Agena-spacecraft separation, 9.05 minutes after start of spacecraft antenna deployment, and 11.2 minutes prior to the initiation of Sun acquisition. DSS-41 acquired the spacecraft two-way 50.1 minutes from launch at a signal level of −94.0 dbm. Mode 4 was in effect 45 minutes from launch by SPC and turned off by RTC at 1 hour and 29 minutes after launch.

Good data from DSS-41 at 49 minutes after launch indicated that the high-and-low-gain antennas and all solar panels were properly deployed. The spacecraft-received signal strength (AGC) and static phase error (SPE) were within the expected command limits.

The ranging receiver was turned on approximately 58 minutes after DSS-41 acquired two-way; 26 minutes later the ranging modulation was turned on and ranging continued for 4.7 hours. At 4 hours, 2 minutes after launch, DSS-51 acquired the spacecraft three-way and continued to track for 4 hours, 27 minutes. DSS-62 acquired three-way at 6 hours, 13 minutes after launch. Station handover from DSS-41 to DSS-62 occurred 1 hour, 14 minutes later with no problems.

The spacecraft began the plus 360-degree roll maneuver 50.3 minutes after launch and completed the maneuver approximately 12 minutes later, immediately prior to Sun acquisition. No problems were experienced with the receiver maintaining rf lock during the maneuvers.

During the cislunar flight, two high-gain-antenna maps were obtained during 360-degree roll maneuvers for star mapping operations. The antenna maps showed that the spacecraft roll position determined by the attitude control subsystem (ACS) and by antenna boresight agree within 2 degrees.

3.2.3 Lunar Injection Through Final Readout

3.2.3.1 Link Performance

Telemetry Link — Downlink telemetry operation was satisfactory throughout the mission. The downlink power level varied with transponder temperature changes as expected, although the temperature excursions did not vary over a wide range. Since the TWTA was on for a major part of the mission, the average signal strength margin for telemetry reception was in excess of 40 db above the nominal link design.

Video Link – Video link performance was satisfactory throughout the mission. Signal levels recorded at the Deep Space Stations during readout varied from −90.6 to −98.9 dbm, which corresponds respectively to video margins of 6.9 db above and 1.4 db below the nominal link design. No readout periods were degraded due to low signal levels from the spacecraft.

3.2.3.2 Computer Program Performance

Both the TRBL program used to determine high-gain-antenna position, and the SGNL program used to determine signal margin, were used successfully throughout the mission.

3.2.3.3 Component Performance

Transponder – Transponder performance was satisfactory throughout the mission. The transponder power output (Telemetry Channel CE10) varied inversely with transponder temperature (Telemetry Channel CT02) as expected.

The range of power and temperature variations was 460.5 mw at 61.7°F (just after launch) to 417.8 mw at 86.9°F (during the final orbit of the prime mission). The transponder high temperature occurred during readout on Orbit 141 when it reached 87.7°F. The transponder power output at this time was 417.8 mw. Typical operation was in the range of 424 to 453 mw at a temperature of 74 to 82°F.

The transponder AGC (Telemetry Channel CE08) indicated normal effect of increasing range on the uplink signal strength during cislunar flight as well as changes in ground transmitter power levels. Command modulation could be observed on CE08 by one tone causing a decrease of approximately 2 db and two tones causing a decrease of approximately 4 db as expected.

During ranging modulation, the uplink carrier power decreased 8 to 10 db as expected.

The transponder static phase error (SPE), Telemetry Channel CE06, displayed a normal sinusoidal cycle variation during the mission. Since the transponder frequency was biased (generally in the negative direction) to avoid interference with the other orbiting spacecraft, the SPE remained positive throughout the most of the mission. During the first part of the cislunar phase and during velocity maneuvers, the transponder was adjusted to best lock frequency and the SPE was effectively at 0 degree.

During DSS handover, if the rising station lost phase lock on the spacecraft signal, they reacquired lock at the predicted spacecraft best-lock frequency, so that SPE would be effectively at 0 degree. Following acquisition, the transmitter was adjusted to the requested offset frequency. This procedure was required only once during the mission. While operating at the offset frequency, the SPE varied from +2.81 to +7.72 degrees. The variations in SPE were attributed to changes in spacecraft received frequency due to doppler and transponder temperature changes. No problems with SPE were noted throughout the mission.

Prior to deboost, DSS-12 turned off ranging in the spacecraft and immediately lost two-way lock. It was reacquired approximately 5 minutes later. This occurrence was not investigated further as ranging was normally discontinued during the photo mission. (During the extended mission it was determined that the ranging loop, was operating properly.)

Multiplexer Encoder – The multiplexer encoder performed satisfactorily throughout the mission. There were no indications of failures or anomalies in the telemetry. All telemetry channels performed satisfactorily, indicating that all channel gates operated properly. The multiplexer encoder zero reference voltage (Telemetry Channel CE01) varied from 0 to +20 mv. The allowable excursion was ± 40 mv; since CE01 was within the expected voltage limits, this indicated that the ground studs were properly grounded.

The command verification word (Telemetry Channel CC03), spacecraft identification (Telemetry Channel CC01), and the external/internal clock indication (Telemetry Channel CC06) were correct throughout the mission.

It was noted that occasionally a random number of consecutive "ones" preceded by a 1010-bit pattern occurred in the command word verification time slot CC03. This is a normal equipment function that occurs when a command is transmitted, if certain other timing events are satisfied.

Command Decoder – The command decoder operated satisfactorily throughout the mission. There were no errors in any of the verified words that were executed into the flight programmer.

Modulation Selector – No problems were experienced in the operation of the modulation selector throughout the mission.

Signal Conditioner – The operation of the signal conditioner throughout the mission was satisfactory. The signal conditioner reference voltage (telemetry channel CE09) varied from 4.72 to 4.78 volts which is within the $\pm$ 1% required.

High- and Low-Gain Antennas – Both the high- and low-gain antennas operated satisfactorily throughout the mission. The antennas were both deployed by stored-program commands following Agena separation. Verification of successful deployment was obtained from discrete Telemetry Channel Measurements CC04 (high-gain antenna) and CC05 (low-gain antenna).

The gains of both antennas were nominal. Based on DSS-received signal levels and the communications system link analysis, the average gain of the high-gain antenna was approximately 24.5 db. and the low gain antenna exhibited a gain pattern similar to that experienced in previous missions and during antenna development.

The high-gain antenna responded successfully to all rotation commands as verified by the telemetered rotation angle (Telemetry Channel CD01). The antenna rotated 20 degrees left and 299 degrees right during the mission. Total antenna movements (including repetition movements) were 36 degrees left and 317 degrees right.

Traveling-Wave-Tube Amplifier – During the mission, the TWTA was commanded on and off six times with a total operating time of 469.63 hours. The TWTA collector temperature, Telemetry Measurement CT01, averaged 164 °F. The maximum temperature reached on CT01 was 171.0°F for a short period of time.

The TWTA power output, Telemetry Measurement CE02, was satisfactory during the entire mission. The power output varied from 11.2 to 11.5 watts, depending on the temperature of CT01.

The TWTA helix current, Telemetry Measurement CE04, was satisfactory throughout the mission. The minimum operating current was 8.04 ma and the maximum was 8.21 ma.

The TWTA collector current, Telemetry Measurement CE05, closely followed CE04, except that its variations were opposite those of CE04 as expected. The values of CE05 were from 42.8 to 43.1 ma.

The TWTA collector voltage, Telemetry Voltage CE03, was normal throughout the mission. The voltage varied between 1,223 volts and 1,228 volts, depending on collector temperature.

3.3 POWER SUBSYSTEM PERFORMANCE

Power subsystem performance was entirely satisfactory throughout Mission V. All components operated within their design requirements and no constraints were imposed on flight operations beyond the requirement that array illumination be sufficient to meet the demands imposed by energy balance requirements.

3.3.1 Power Subsystem Configuration

The power subsystem configuration described in preceding reports applies to that used in Mission V, except that a booster regulator was provided in Mission V to maintain 30.5 volts (nominal) to the photo subsystem regardless of the degree of solar panel illumination. Thus, two qualities of bus voltage (in addition to squib bus voltage) were provided; (1) spacecraft bus voltage, which was limited to 30.56 volts by the shunt regulator, but decreased to the level of battery voltage during absence of solar panel illumination; and (2) camera bus voltage, which was maintained at 30.5 volts continuously. The input to the booster regulator is the spacecraft bus voltage; its output is the camera bus voltage. The booster regulator load is the photo subsystem, including its heaters. This configuration enabled operation of the photo subsystem in the "camera" mode on nominal 30.5 volts regardless of the magnitude of the total angle between spacecraft roll axis and the Sun vector. Mission V design called for large total angles for each photo maneuver, 13 of which were over 90 degrees.

3.3.2 Launch to Acquisition

At 6 minutes prior to liftoff, the spacecraft was put on internal power. From this time until Sun acquisition, all electrical loads were supported by the spacecraft battery. During the 42 minutes of discharge visible to Station 71, the average load was 4.37 amps and battery voltage had decreased to 25.12 volts. It is estimated that the battery discharged to a depth of 25%. When Station 41 rose, panels had deployed and the battery was charging at the maximum rate of 1.35 amps. The solar array was supplying 12.47 amps at 30.56 volts.

3.3.3 Cislunar Flight to Lunar Injection

During this period, the on-Sun array current remained constant at 12.43 amps. Battery charge was reduced as a function of voltage and temperature. Early in the cislunar phase, battery temperature was 55 to 60°F and charge reduction (taper) started when battery voltage increased to 30 volts. Later in the cislunar flight, with battery temperature at 95°F, taper started at 28.5 volts. At injection, battery temperature had reached 101°F and voltage did not get high enough to initiate taper charge.

At the midcourse maneuver, the heavy engine solenoid load was supported by the array with no impact on bus voltage. Other significant load profile increments occurred August 2 when heater power and the TWTA were turned on, and August 3 when the TWTA was turned off. Table 3-1 shows electrical load values.

Spacecraft attitude during the injection altitude maneuver enabled 2 amps of assistance from the array during the engine burn, so that the minimum bus voltage was 24.16 volts and the battery was discharged only to a depth of approximately 10%.

3.3.4 Initial Ellipse

This period provided the first Mission V inflight tests of several power subsystem components. During photo maneuvers, the load was supported entirely by the battery, and the camera bus voltage was held at 30.52 volts by the booster regulator. Readout and processing were accomplished at spacecraft attitudes that allowed less than 1 amp of shunt regulator current, with spacecraft bus voltage held constant at 30.56 volts.

The constant-voltage mode of charge controller operation was demonstrated after photo maneuvers when the battery-cooling effect of the discharge combined with the increasing battery voltage to initiate a reduced charge rate.

During this period, the heavier photo subsystem loads associated with readout and processing (see Table 3-1) were first supported.

At the time of maneuver for first-orbit transfer, battery temperature had increased to 104°F. During the maneuver, the electrical load was shared between the array and battery, so that discharge depth was shallow (8%) and taper charge was initiated only 21 minutes after the maneuver.

3.3.5 Intermediate Ellipse

Photo maneuvers performed during this period (and prior to Site A-14) required that the battery support the photo environmental heater load when "heater power on" was commanded immediately after the camera exposure. With the spacecraft cooling during the off-Sun maneuver, photo heater thermostats were call-

Table 3-1: Electrical Loads

Operational Mode	Photo Heaters: Inhibited	Photo Heaters: Solar Eclipse On	Photo Heaters: Solar Eclipse Off	Spacecraft Loads EE07 (amps): Min	Nom	Max	Power Source
Cislunar	X			3.43	3.62	3.68	Power Array Supplying Battery Charging
Cislunar		X		3.68	4.25	4.49	
TWTA on	X			--	5.29	--	
Engine burn		X		--	9.01	--	
Photo standby and TWTA			X	6.68	6.86	7.10	
Processing and TWTA			X	5.96	7.75	8.05	
Readout	X			6.86	6.92	7.10	
Engine burn		X		8.17	8.59	8.83	Battery Supplying Power
Photo maneuver and TWTA	X			5.29	5.90	6.02	
Photo sequence and TWTA	X			5.96	6.02	7.22	

ing for heat as soon as enabled. Thus, battery loads were initially high and bus voltage decreased to 23.84 volts in one case (Site A-7.1). Load current approached 8 amps. This situation was later remedied by changing the time relationship of heater commands with maneuver commands (see "Final Ellipse"). Except for about 5 hours during this period when the spacecraft was pitched 39 degrees off the Sun, it was operated on-Sun and battery temperatures reached 108°F at times.

The transfer maneuver to final ellipse discharged the battery approximately 10% and the 8 amp load during engine burn reduced bus voltage to 23.68 volts.

3.3.6 Final Ellipse through Photography

During this period, minimum bus voltage during photography was increased by disabling photo heaters before misorienting the array from the sunline and enabling the heaters after reorienting. With this system, the minimum bus voltage observed was 24.32 volts, and the average minimum for all photo maneuvers was 25.22 volts.

Readout and processing was supported at off-Sun angles up to 41 degrees. The minimum shunt regulator current observed was 0.63 amp.

3.3.7 Final Readout

The remainder of the power subsystem mission was used to support photo readout. The nominal mode of operation was with the spacecraft operating 40 degrees off-Sun, load current 6.98 amps, array current 9.49 amps, shunt regulator current 1.05 amps, and battery charge rate 1.35 amps.

3.3.8 Component Performance

Solar panel performance was entirely nominal and has been documented in preceding reports. It varied from preceding missions only as a result of solar environment unique to Mission V. As in Mission IV, the spacecraft was constantly illuminated. Unlike Mission IV, the solar intensity increased approximately 1% during the mission, thus tending to conceal array degradation during this period. Temperature instrumentation was limited to two panels (1 and 4). Electrical performance, as indicated by telemetry, was within the design requirements of the array current was approximately 200 ma less than predicted attributed to Telemetry Channel EE03 calibration. This channel was apparently near its lower limit of tolerance, because the sum of all spacecraft electrical loads and losses totaled approximately 200 ma greater than the indicated array current; this relationship prevailed throughout the mission. Indicated array current with the panels normal to the solar vector varied only from 12.49 amps at initial acquisition to 12.37 on August 28 at completion of photographic mission. This decrease, combined with the previously mentioned increase in solar intensity, indicates array degradation of approximately 2%.

Battery and Charge Controller — The maximum charge current for Mission V was set at 1.35 amps. This constant-current mode of operation was predominant throughout the mission. The constant voltage mode (overcharge) was initiated in accordance with the voltage-temperature characteristic shown as Figure 3-1. The charge rate never dropped below 1.0 amp when this mode was used. Battery temperature never exceeded 112.9°F, so the trickle charge mode (initiated by the charge controller at 125°F) was never used.

Mission design was such that during each photo maneuver, the battery was either sharing the electrical load with the array or supplying the total power requirement. During these maneuvers, the minimum observed bus voltage was 23.84 volts and the maximum depth of discharge was 11.5% at Photo Site A-7. Bus voltage and battery current are shown plotted in Figure 3-2 for Photo Site A-22, selected as representative.

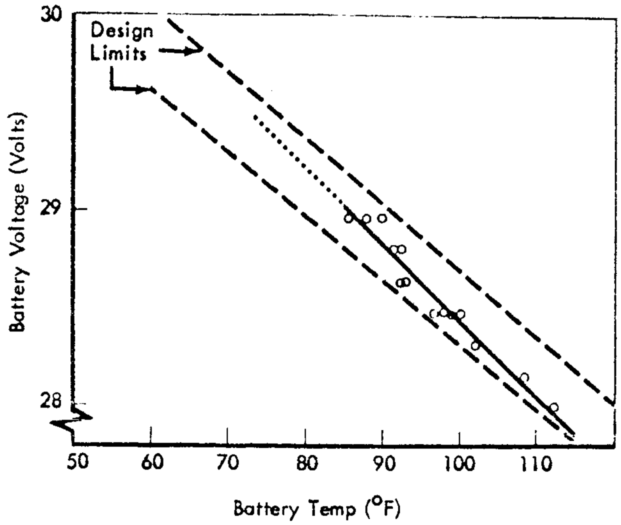


Figure 3-1: Battery Overcharge Voltage vs Temperature

Maximum depth of discharge during the mission was approximately 25%, occurring at initial Sun acquisition; minimum bus voltage for the mission was 23.68 volts during the second-ellipse transfer.

Performance of both battery and charge controller were entirely nominal throughout the mission.

Shunt Regulator — This unit limited the spacecraft bus voltage to 30.56 volts throughout the mission. Included within the unit is circuitry associated with providing voltage, current, and temperature telemetry. All performed without difficulty.

Booster Regulator — This unit provided a constant potential of 30.52 ± 0.16 volts to the camera bus throughout the mission. Input voltage varied from 23.68 to 30.56 volts. The conversion efficiency of this regulator was 75 to 80%; thus, the cost of operating it in terms of load current was approximately 0.25 to 0.35 amp during camera, processing, or readout activity.

3.4 ATTITUDE CONTROL SUBSYSTEM PERFORMANCE

3.4.1 Attitude Control Subsystem Configuration

The attitude control subsystem maintains spacecraft attitude control with respect to

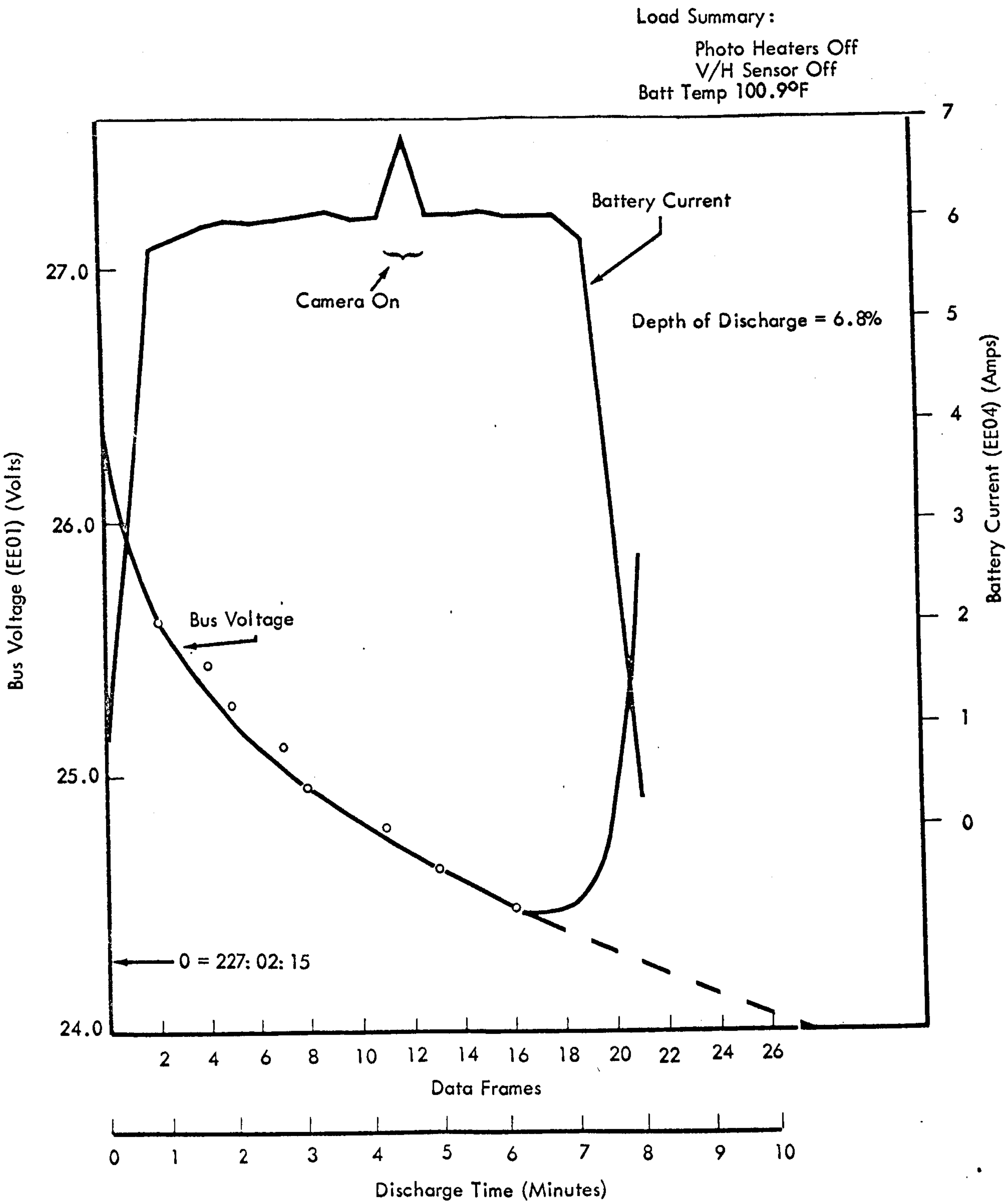


Figure 3-2: Battery Discharge – Photo Site A-22, Orbit 54, Day 227

inertial and celestial references. Control with respect to celestial references (celestial hold) is accomplished using sun sensors in the pitch and yaw axes and a Canopus tracker in the roll axes for position reference. Rate damping is provided by a single axis floated gyro in the rate mode on each axis. Spacecraft control with respect to inertial reference (inertial hold) is by means of the gyros in the rate integrating mode for all three axes. Lead-lag networks on the output of the byros are used for rate dumping. Maneuvers are performed with the gyros in the rate mode. Integration of rate mode output is used to measure and control maneuver angles. Control torques are generated by nitrogen thrusters located on the engine mount deck. Control of pitch and yaw attitude during engine burns is by means of actuators that vector the engine in response to rate integrating mode output of the gyros. Throughout the mission, the attitude control subsystem maintained stable operation for both reaction control and thrust vector control.

The Lunar Orbiter V attitude control subsystem performed with sufficient accuracy to satisfy all mission objectives. Spacecraft control techniques that were developed during the previous missions were used successfully during Mission V. Thermal constraints and degradation of the thermal coating on the equipment mounting deck required an off-Sun attitude for approximately 66% of the mission. The Canopus tracker was used in an open-loop mode throughout the mission. Nitrogen gas was conserved by biasing initial roll maneuvers to cancel any Canopus reference error and by biasing final pitch maneuvers to place the spacecraft in a pitched-off attitude to meet thermal constraints.

3.4.2 Attitude Control Subsystem Performance
During Mission V, the attitude control subsystem (ACS) performed its many design tasks within specification. The ACS performed 472 maneuvers during the photo mission, which ended at Day 240, 02:00:00 GMT. Attitude maneuver rates for all axes were within the design limits of 0.55 ± 0.05 degree per second for maneuver in narrow deadband. The maneuver rate in the wide deadzone was 0.063 degrees per second in pitch (only one pitch maneuver was performed in the wide deadzone). The ACS maintained spacecraft orientation with respect to the Sun and Canopus on command within ± 0.2 and ± 2.0 degrees, depending on the selected deadband. Deadband accuracies were within telemetry resolution for narrow and wide deadzones.

Attitude control was maintained with the spacecraft pitched from 30 to 54 degrees away from the Sun for approximately 66% of the mission. The pitched-off attitude was required to maintain spacecraft thermal balance and delay paint degradation. Drifts in inertial reference were within design limits, which reduced the frequency of updating this reference.

Stable thrust vector control of spacecraft attitude was maintained through four velocity control engine burns. Based on telemetry data, engine-burn termination was performed by the accelerometer within design tolerance.

Due to glint problems associated with the Canopus tracker, the roll axis was never put into the conventional limit cycle mode using Canopus roll error in closed loop.

Operational methods used to control spacecraft attitudes by mission phases to meet mission requirements are presented below.

Cislunar Coast — Throughout the cislunar coast phase the spacecraft remained on-Sun except for the midcourse maneuver and drift tests. Canopus was not acquired in a closed loop during the cislunar phase of the mission to prevent possible loss of the reference star due to tracker glint. Canopus reference was maintained by rolling the spacecraft to place Canopus in the tracker field of view.

Velocity Change Maneuvers — Attitude maneuvers for midcourse correction, lunar orbit injection, and the two orbit transfers were started from a closed-loop Sun reference in the pitch and yaw axes. The roll reference for the attitude maneuvers was obtained by calculating the expected angle between Canopus and the gyro null position at the time the roll maneuver for the burn was executed. The stored roll maneuver was then adjusted by the amount of the expected roll error at the time of the maneuver. The roll axis remained in inertial hold throughout the mission. Estimated roll errors for the start of the velocity change attitude maneuvers were +0.01, −0.004, −0.08, and +0.12 degree for first midcourse, lunar orbit injection, first transfer, and second transfer, respectively.

Photo Maneuvers — Maneuvers for all photographs were started from a closed-loop reference on the Sun in pitch and yaw. The roll reference was updated by overstoring the programmer as described above. The roll error was usually measured near the lunar South Pole. Just prior to the lunar South Pole, the spacecraft acquired the Sun from the thermal pitch-off attitude and the roll error was measured by cycling the Canopus tracker on and off. The roll maneuver for photography was adjusted by the roll error at the measurement plus the expected gyro drift between the measurement and the time for the roll maneuver. The spacecraft remained on the Sun until the photographic maneuvers were begun for photos on the perilune side of the Moon. When the photos were on the apolune side of the Moon, a thermal pitch-off Sun maneuver and a Sun acquisition were performed between the measurement of roll error and the start of the photo maneuvers.

All photographs on the perilune side of the Moon were taken using three-axis maneuvers. The first forward maneuver was always a roll maneuver so that roll error could be corrected. The last return maneuver was always a pitch maneuver, which was biased to return the spacecraft to a pitched-off-Sun attitude.

Photo maneuvers on the apolune side of the Moon were usually two-axis maneuvers. A thermal pitch-off maneuver was usually performed following return from the photographic attitude.

Readout — The final readout phase of the mission was performed in a pitched-off-Sun attitude. Pitch and yaw attitudes were monitored using solar array current and yaw coarse Sun sensors as in previous missions. The roll attitude was monitored by cycling the tracker on and off once per orbit. Update maneuvers were performed as needed to maintain spacecraft attitude within thermal, power, and antenna pointing constraints. For final readout, the spacecraft remained in the off-Sun attitude for 8 days, 10.5 hours.

3.4.3 Component Performance

Canopus Star Tracker — The Canopus tracker was first turned on at Day 214, 05:21 GMT, approximately 7 hours into cislunar flight. Initially, it was in the track mode indicating a roll error minus 3.2 degrees and a map signal of 1.12 to 1.18 volts. This star was later identified as Acrux (α Crucis). The star was tracked for approximately 2 minutes, at which time the roll error went to positive saturation due to glint. At Day 214, 05:30 GMT, a plus-360-degree roll maneuver was performed to obtain a star map. Star map voltage remained relatively constant throughout the maneuver, with the tracker locked on positive glint. The map contained no significant information.

While waiting to make an antenna map, a roll drift test was performed using Acrux as a reference star. It was difficult to maintain track on Acrux because the star was near the edge of the tracker field of view. Successful tracks were obtained about half the time. At Day 214, 10:14 GMT, a minus-357-degree roll maneuver was performed to place Acrux in the center of the field of view to improve the chances of tracking the star when the tracker was cycled. The negative direction was selected to pull the tracker off Acrux into negative rather than positive saturation so that a star map could be obtained. In this map, two small pips occurred, that were later shown to correspond with Canopus and Mars. at Day 214, 14:52 GMT, the TWTA was turned on and a roll-minus-360-degree maneuver was performed, resulting in a third star map and first antenna map. The DSS indicated peak signal strength occurred at Day 214, 15:03:50 GMT at a roll angle of −352.5 degrees. TRBL showed a roll plus 49.5 degrees from Canopus to maximum signal strength. This led to the conclusion that Acrux was the star being tracked and the spacecraft should be rolled minus 42 degrees to Canopus. Calculations from the star map data using the blip at 52.5 degrees as Canopus and accounting for the 9-degree error due to lockup on negative glint indicated about 43.5 degrees would be a better angle. The *a priori* confirmed this angle between Acrux and Canopus.

At Day 214, 18:24 GMT, the spacecraft was rolled minus 43.5 degrees and Canopus was tracked. The map signal was 2.3 to 2.4 volts and the roll error was minus 0.150 degree. This was followed by a second antenna map, which indicated maximum signal strength at plus 44.0 degrees from Canopus. This agreed with the TRBL run for Day 214, 18:30 GMT. Since it was a roll plus, the tracker went to positive roll error saturation and locked on glint so there was no useful information in the map. At the completion of the plus-360-degree roll, the roll error was minus 0.165 degree.

Canopus was tracked without acquisition for approximately 4 hours prior to both midcourse and injection. For the remainder of the photo mission, the basic operational procedure was to turn the tracker on without acquisition once or twice an orbit to obtain a roll reference and update prior to a photo sequence. On those occasions when the tracker was locked on glint, track was usually regained by performing an off-on cycle.

Approximately 9 hours prior to the end of the basic Lunar Orbiter V mission (Day 239, 17:14 GMT), the deadzone was opened and drift tests in pitch, roll, and yaw were started. On three occasions thereafter (Day 239, 17:44, 18:18, and 20:25) the tracker was turned on and the bright-object shutter was closed. Analysis of this data revealed that this was caused by the tracker's extreme sensitivity to glint as a function of yaw attitude. Each time the spacecraft was on the negative side of −0.75 degree from the Sun, the BOS closed, which corresponds to the tracker centerline moving from 84 degrees (Sun angle) to an angle of 83.25 degrees.

The star map signal was initially 2.4 volts, decayed to 2.1 volts and recovered to approximately 2.9 volts by the end of the mission.

Through Day 240, 02:00 GMT, the tracker had been on for a total of 34 hours, 10 minutes. It had been cycled on and off 198 times, seven of which were required because of glint problems.

Flight Programmer – Flight programmer performance during Mission V was nominal. As of August 27, 1967, the programmer had processed 1,703 real-time commands, stored in memory 2,822 stored-program commands, and executed approximately 14,000 commands from its stored-program routine. Total clock error was +0.66 second, reflecting a 1.23-millisecond-per-hour drift rate.

Sun Sensors – The sun sensors performed without difficulty for Mission V, providing a celestial reference for a variety of situations.

Initial Sun acquisition took place automatically within the required 60 minutes from launch. During the initial Sun acquisition, as soon as the telemetry data was "good" (50 minutes after launch), it was observed that the Sun had already been acquired in pitch and yaw. The exact time of acquisition could not be determined. Reacquisition of the Sun after Sun occulation or attitude maneuvers was performed approximately 91 times; 89 of these acquisitions were done in the narrow deadband and two were done in the wide deadband. Every acquisition went as expected.

The capability of switching between fine, coarse and fine, and coarse-only sun sensors proved invaluable for off-Sun operation (66% of mission time was spent off-Sun). The ability to stay off-Sun for extended periods of time using the coarse sun sensors greatly reduced nitrogen consumption. Yaw sun sensor degradation due to a large pitch attitude was approximately the same as observed for previous missions. This degradation at a pitch angle of 40 degrees is approximately 0.58. Moonlight on the coarse sun sensors caused shifts in error output for various portions of the orbit as on previous missions; these shifts had no effect on the mission.

Closed-Loop Electronics – The closed-loop electronics which performed without difficulty throughout the mission, selected — on command from the flight programmer, the inertial reference unit, the sun sensors, and the Canopus star tracker — loop closure between sensor outputs and vehicle dynamics. As in past missions, the crab angle sensor modes were not used in conjunction with the attitude control subsystem.

There was no opportunity to check yaw sun sensor limits or the pitch minus sun sensor limit because the photo maneuvers were performed with only the fine sun sensors on. The pitch plus sun sensor had a "hard" limit at 30 degrees.

The minimum-impulse circuit appeared to be operating between 11 (nominal) and 14 milliseconds. Approximate single pulses were experienced 30, 70, and 50% of the time during limit cycle operation for the roll, pitch, and yaw axes, respectively. Additional pulsing data are contained in the reaction-control section of this report.

Compensation networks for the thrust vector control subsystem and inertial reference unit compensation network (lead-lag) was used to obtain pseudorate whenever the gyros were in the inertial hold mode. Performance by this network during attitude holding operations was very similar to that obtained with the gyros in the rate mode.

Reaction Control Subsystem — The reaction control subsystem thrusters performed satisfactorily during the mission. The number of thruster operations for the photographic mission was estimated by reviewing vehicle telemetry data for the mission, and are tabulated below.

Individual thruster performance was evaluated for as many of the spacecraft maneuvers as possible. Actual, predicted, and specification values for each axis are tabulated below.

Thruster Operations

Mode	Roll	Pitch	Yaw	Total
Limit Cycle	4,100	5,740	6,150	15,990
Maneuvers	344	491	291	1,126
	4,444	6,231	6,441	17,116

Thruster Performance

Axis	Actual Thrust (lbs)	Predicted (lbs)	Specification Value (lbs)
Roll	0.070 ± 0.004	0.061 ± 0.003	0.051 to 0.070
Pitch	No Data	0.058 ± 0.003	0.045 to 0.062
Yaw	0.069 ± 0.007	0.054 ± 0.003	0.045 to 0.062

The observed thrusts are higher than the predict values. The yaw and roll thrust values are within specification tolerance but do tend to the high side. These slightly high thrust values in no way degraded the mission. Also, maneuver accuracy was not degraded and nitrogen consumption was not increased.

Slight cross coupling was observed during maneuvers and limit cycle; however, in view of the data observed, it is impossible to estimate cross-coupling magnitude or even to determine if it is caused by thruster misalignment or gyro cross coupling.

Thrust Vector Control – **Control of the spacecraft attitude during the four engine burns was performed within specification by the thrust vector control system. Residual spacecraft rates after each burn were lower than predicted maximus for stable TVC limit cycle operation.**

Lateral travel of the c.g. was low for this spacecraft, as indicated by actuator travel. Maximum actuator excursion for the burns was 0.21 degree for pitch (during injection) and 0.43 degree for yaw (during midcourse).

Inertial Reference Unit Summary – The inertial reference unit (IRU), Serial Number 110, was installed on Spacecraft 3 for Mission V. This IRU contained Sperry SYG-1000 gyros, as did the IRU's for Missions I and II. Low and stable drift rates again contributed to efficient gas usage and ease of attitude maintenance during the large portion of time in which all three axes were in inertial hold. As on the last mission, the roll axis was not operated in the rate mode except during maneuvers.

Lab and space drift values are tabulated below. Sign reversals from laboratory to space values have been explained in earlier inputs.

① Two-axis test prior to star map
② Three-axis test
③ Roll test only; pitch and yaw in CLC
④ Three-axis test; insufficient data for roll.

Except for ③ , all drift tests were performed with all three axes in inertial hold. With all three spacecraft axes open loop, there is some drift coupling between axes. Therefore, the drift rates determined cannot be compared directly with the gyro drifts obtained in the laboratory. Also, the drift rates in the laboratory have an opposite sign from those taken in space due to the test methods employed.

Maneuver accuracy data is available for the roll axis only. The maneuver error includes both gyro and flight programmer errors.

Day 214 – 18 (roll +360 degrees) −0.065%
Day 214 – 14 (roll −360 degrees) −0.029%

These maneuver errors are well within the mission tolerance of ± 0.32%.

3.4.4 Nitrogen Consumption

Nitrogen consumption for the attitude control subsystem for Mission V is presented in Figure 3-3.

Rate-Integrate Mode Drift at Null

AXIS	LAB VALUES		SPACE VALUES				
	12/29/66	6/14/67	8/1/67 ①	8/2/67 ②	8/3/67 ②	8/4/67 ③	8/9/67 ④
Roll	−0.015 deg/hr	−0.031 deg/hr	---	+0.045 deg/hr	+0.038 deg/hr	+0.064 deg/hr	---
Pitch	−0.065 deg/hr	−0.093 deg/hr	+0.075 deg/hr	+0.078 deg/hr	+0.094 deg/hr	---	+0.094 deg/hr
Yaw	−0.043 deg/hr	−0.038 deg/hr	+0.071 deg/hr	+0.093 deg/hr	+0.090 deg/hr	---	+0.117 deg/hr

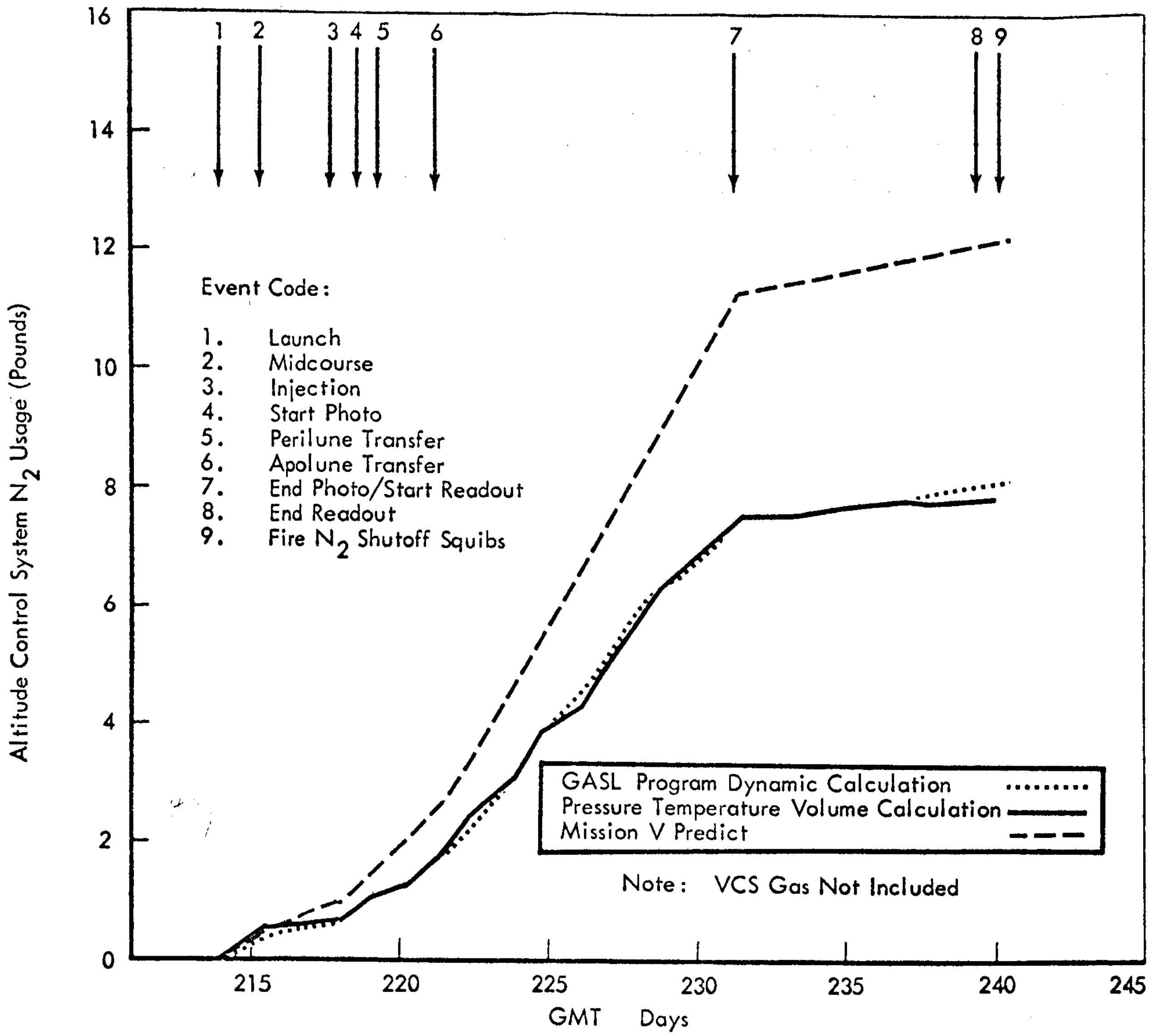


Figure 3-3: Attitude Control Subsystem Nitrogen Usage

Mission V exhibited the following characteristics that led to low nitrogen usage, as on previous missions.

- No major problems with the star tracker in locating Canopus.
- The operational methods used to fly off the Sun eliminated a large number of pitch maneuvers.
- No leakage problems occurred in the VCS as in Mission I.
- Lower gyro drift rates minimized the number of update maneuvers.

At Day 240, the amount of unaccounted-for nitrogen since launch at Day 213 was approximately 0.20 lb (this is 0.007 lb per day).

The nominal mission predict for N_2 usage was approximately 4.1 lbs. above actual usage at the end of the mission on Day 240. The reasons for this difference follow.

- The use of worst-case values for Sun acquisition in the predict.
- There were fewer attitude updates and thermal pitch-off maneuvers than predicted.
- Slightly smaller gas usage for maneuvers in the end burn condition than predicted.

The total nitrogen used for Lunar Orbiter photo missions follows.

	Mission I (lb N_2)	Mission II (lb N_2)	Mission III (lb N_2)	Mission IV (lb N_2)	Mission V (lb N_2)
Attitude control subsystem	7.80	5.50	7.00	10.50	8.10
Velocity control subsystem	3.38	3.14	2.70	2.65	3.29
Leakage	1.13	0.0	0.0	0.0	0.0
Total	12.31	8.64	9.70	13.15	11.39
Initial N_2 at launch	15.10	15.15	15.17	16.70	16.80
N_2 for extended mission	2.79	6.51	5.47	3.55	5.41

The maneuvers performed on Mission V for launch through Day 240, 02:00 GMT, are given below.

	ROLL		PITCH		YAW		TOTALS	
Purpose of Maneuver	NDZ	WDZ	NDZ	WDZ	NDZ	WDZ		Predicted
Star map & other	6	0	4	0	1	0	11	9
Attitude update	11	0	10	0	6	0	27	43
Thermal pitch-off	0	0	22	1	0	0	23	62
Velocity change	8	0	8	0	0	0	16	16
Photo maneuver	147	0	155	0	93	0	395	402
Subtotal	172	0	199	1	100	0	472	532
Total	172		200		100		472	532

Celestial Acquisitions	NDZ	WDZ	TOTAL
Canopus acquisitions	0	0	0
Sun acquisitions	89	2	91
Deadband closures	4		

WDZ = wide deadzone NDZ = narrow deadzone

3.4.5 Non-Nominal Operations

Conditions were encountered during Mission V that resulted in non-nominal operation of the attitude control subsystem; these are summarized below.

Thermal Conditions – Again on Mission V, spacecraft heating was encountered that required operating the spacecraft in a "pitch-off-Sun" attitude for approximately 66% of the mission. As a result of the pitch-off-Sun requirement, 39 maneuvers were required for thermal pitch-off and updates of the inertial reference as compared to 151, 25, 73, and 78 maneuvers for Missions I, II, III, and IV, respectively.

Tracker "Glint" – Tracker glint was again experienced on Mission V. On seven occasions during Mission V, Canopus track could not be obtained until the tracker was cycled off and on. During a wide-deadzone drift test after final readout, the bright-object-sensor shutter remained closed when the yaw position exceeded −0.75 degree in the negative direction with respect to the Sun, which illustrates how sensitive the tracker is to yaw position. The bright-object-sensor shutter closure did not effect mission performance.

3.5 VELOCITY CONTROL SUBSYSTEM

The velocity control subsystem is a liquid-bipropellant, pressure-fed, bladder-expulsion propulsion system employing a single 100-pound-thrust, radiation-cooled, gimbal-mounted rocket engine for all inflight velocity change maneuvers. The propellants are nitrogen tetroxide and a 50-50 mixture of hydrazine and UDMH; nitrogen gas is the pressurizing medium.

3.5.1 Operation and Performance Summary

Operation and performance of the velocity control subsystem during Mission V was excellent. Four propulsive maneuvers were conducted in support of the primary mission; these were 29.75-mps midcourse maneuver, 643.04-mps injection maneuver, a 15.97-mps perilune transfer, and a 233.66-mps apolune transfer. Premission planning required implementation of two transfer maneuvers; photography was accomplished in the initial, intermediate, and final orbits.

Prelaunch propellant and nitrogen servicing operations were accomplished without difficulty. There were 276.24 pounds of propellant loaded, as were 16.8 pounds of nitrogen; the spacecraft launch weight was determined to be 864.56 pounds. Based on these weight data, the nominal velocity increment capability of the VCS was determined to be 1,015.8 mps with a 3-sigma tolerance of ± 29 mps. This estimate reflects a 99% expulsion efficiency.

Flight data performance analysis indicates that, during the injection maneuver, the most typical, the average delivered thrust was 100.1 pounds. The engine specific impulse was determined to be approximately 276 seconds during all maneuvers. A total velocity change of 922.42 mps has been imparted to the spacecraft with a total engine operating time of 687.9 seconds.

System temperatures were generally in the region of 50 to 75°F – a satisfactory operating regime. Temperature values were in general agreement with those observed during Mission IV, though slightly higher as a result of closer proximity to the lunar surface.

Propellant and nitrogen isolation squib valves were actuated without incident. Prior to actuating the propellant squib valves, the "propellant line bleed" event was successfully conducted. There were no requirements for propellant heater operation during the mission.

Gimbal actuators performed according to expectations. During maneuvers, the pitch actuator varied between −0.06 and −0.25 degree, while the yaw actuator was generally +0.25 to +0.45 degree. All performance was consistent with previous mission results.

All VCS tests conducted during simulated and actual countdowns were accomplished without incident. Each test consisted of a simulated 240-fps maneuver to deflect the gimbal actuators, followed by a 120-fps maneuver to recenter them.

The following sections present the various aspects of the velocity control subsystem's operation during the flight of Lunar Orbiter V

Spacecraft as supported by SPAC at the SFOF. This includes discussion of countdown and flight events, with particular emphasis placed on flight operations and performance of the VCS during propulsive maneuvers. Table 3-2 briefly summarizes the results of the four propulsive maneuvers.

ESA Spacecraft Fueling Operations – After completing all Hangar S checkout tests, Spacecraft was transferred to the Explosive Safe Area (ESA) for fueling, pressurization, further testing, and encapsulation into the nose shroud. The propellant and nitrogen servicing AGE functioned without difficulty. Table 3-3 summarizes the servicing operations that took place on July 15, 1967. It will be noted that the amount of nitrogen is consistent with Mission IV and greater than that on the initial three missions; this follows from the extra amount of gas required for attitude maneuvering for additional photographic activity.

Table 3-2: Maneuver Performance Velocity Control Subsystem

	Velocity Change (mps)	Burn Time (sec)	Thrust (lbs)	Specific Impulse (sec)
Midcourse				
Predict	**29.76**	**26.4 ± 0.08**	99	**276**
Actual	**29.75**	**26.1**	99.9	**276**
Injection				
Predict	643.04	501.1 ± 8.5	99.5	276
Actual	643.04	498.1	100.1	276
Perilune Transfer				
Predict	15.97	10.95 ± 0.6	100	276
Actual	15.97	10.78	101.6	276
Apolune Transfer				
Predict	233.67	151.1 ± 4	101	276
Actual	233.66	152.9	99.9	276

Table 3-3: Propellant and Nitrogen Servicing Summary - - Velocity Control Subsystem

	Fuel	Oxidizer	Nitrogen
On-board, lbs.	94.30	181:94	16.8
Ullage volume, cu in	70.52	126.8	- -
Pressure, psig	45	45	3990
Temperature, °F	58	56	58

After completion of the VCS servicing, the complete flight-configuration spacecraft was weighed and balanced; launch weight was determined to be 864.56 pounds. Calculations were performed to ascertain the velocity increment capability of the spacecraft based on the aforementioned weights and available rocket engine performance. The "Delta V" capability was found to be 1,015.8 ± 29 meters per second.

Operational Readiness Test - - The formal operational readiness test (ORT-1) was initiated on July 28 (Day 209); with first telemetry at 14:11 GMT all VCS parameters were normal and in a "go" condition. The VCS countdown test was initiated at 17:22 GMT, the first maneuver deflecting the pitch actuator to −0.889 degree and the yaw actuator to +0.245 degree. The actuators were recentered at 17:27 GMT; maximum engine valve temperature was 74.3°F. All events proceeded satisfactorily to a simulated liftoff at 21:30 GMT.

Launch and General Mission Events -- Launch countdown was initiated on August 1, 1967 (Day 213) with first telemetry at 12:07 GMT; all VCS parameters were normal. The VCS countdown test was successfully conducted at 15:19 GMT, resulting in pitch and yaw actuator deflections of -0.912 and +0.245 degree, respectively; maximum engine valve temperature was 82.0°F. Vehicle liftoff occurred at 22:33:00.338 GMT on Day 213 after approximately a 1-hour "hold" for inclement weather at ETR. Real-time telemetry loss was minimal until acquisition of the spacecraft by DSS-41 at 23:22 GMT; the spacecraft separated from the Agena at 23:08:33.8 GMT.

Upon acquisition by DSS-41, it was verified that the propellant tank had been pressurized to values of 189.5 and 189.1 psia, fuel and oxidizer, respectively. This was a little lower than the first three missions, but consistent with the regulator characteristics and Mission IV values.

The next significant VCS event concerned bleeding the propellant lines and arming the system. The "bleed" event occurred at 14:23 GMT on August 2 (Day 214); the engine valves were open for approximately 30 seconds, thereby increasing the valve temperature by 10.5°F. This activity was followed by propellant squib valve actuation at 14:28 GMT; propellant tank pressures decayed 2 psi, providing positive confirmation of valve actuation.

The midcourse maneuver for trajectory adjustment was designed for engine ignition to occur at 06:00:00.0 GMT on August 3 (Day 215) with a desired velocity change of 29.76 mps. The maneuver was conducted without incident, a velocity change of 29.75 mps being achieved with an engine operating time of 26.1 seconds.

The orbit injection maneuver was programmed for engine ignition to occur at 16:48:54.4 GMT on August 5 (Day 217); the desired velocity change was 643.04 mps. The maneuver produced orbital elements very close to the desired values. Engine operating time was determined to be 498.1 seconds; engine valve temperature was 68 to 71°F during engine operation, and reached a maximum value of 109.8°F approximately 53 minutes following the maneuver.

The mission was designed on the basis of conducting two orbit transfer maneuvers: the first was a perilune transfer maneuver to lower the perilune to approximately 100 kilometers, and the second was an apolune transfer maneuver to lower the apolune to approximately 1,500 kilometers. The perilune transfer maneuver was initiated at 08:43:48.7 GMT on August 7 (Day 219) with a programmed velocity change of 15.97 mps; the desired velocity change was achieved with an engine operating time of 10.78 seconds. The perilune of the resulting orbit missed the desired value by less than 0.5 kilometer, an error of less than 0.5%. The apolune transfer maneuver was programmed for an engine ignition at 05:08:32.7 GMT on August 9 (Day 221) with a desired ΔV of 233.67 mps. Again, the maneuver was conducted without incident, a velocity change of 233.66 mps being achieved with an engine operating time of 152.9 seconds. From tracking data, the error in the achieved apolune altitude was less than 0.04%.

At initiation of extended-mission operation on August 28 (Day 240), Spacecraft had a remaining velocity change capability of approximately 93 mps, and approximately 5.6 pounds of nitrogen for attitude control purposes. The nitrogen shutoff squib valve was actuated at 22:38 GMT on August 27 (Day 239).

Subsystem Time-History Data -- Figure 3-4 presents the quantity of nitrogen gas remaining in the storage vessel as a function of time throughout the mission. The data points are plotted at 6-hour intervals and represent a 6-hour average value. For reference, a nominal mission budget and a significant-events code are included in the plot. It will be observed that actual nitrogen consumption was quite close to the predict for initial mission phases, but then began to deviate (in a favorable direction) during the photographic phase. The deviation is explained as follows.

- The budget included allowances for update roll attitude errors. In actual fact, no physical roll maneuvers were conducted; rather, the roll error was overstored into the initial attitude maneuver for the photographic site.

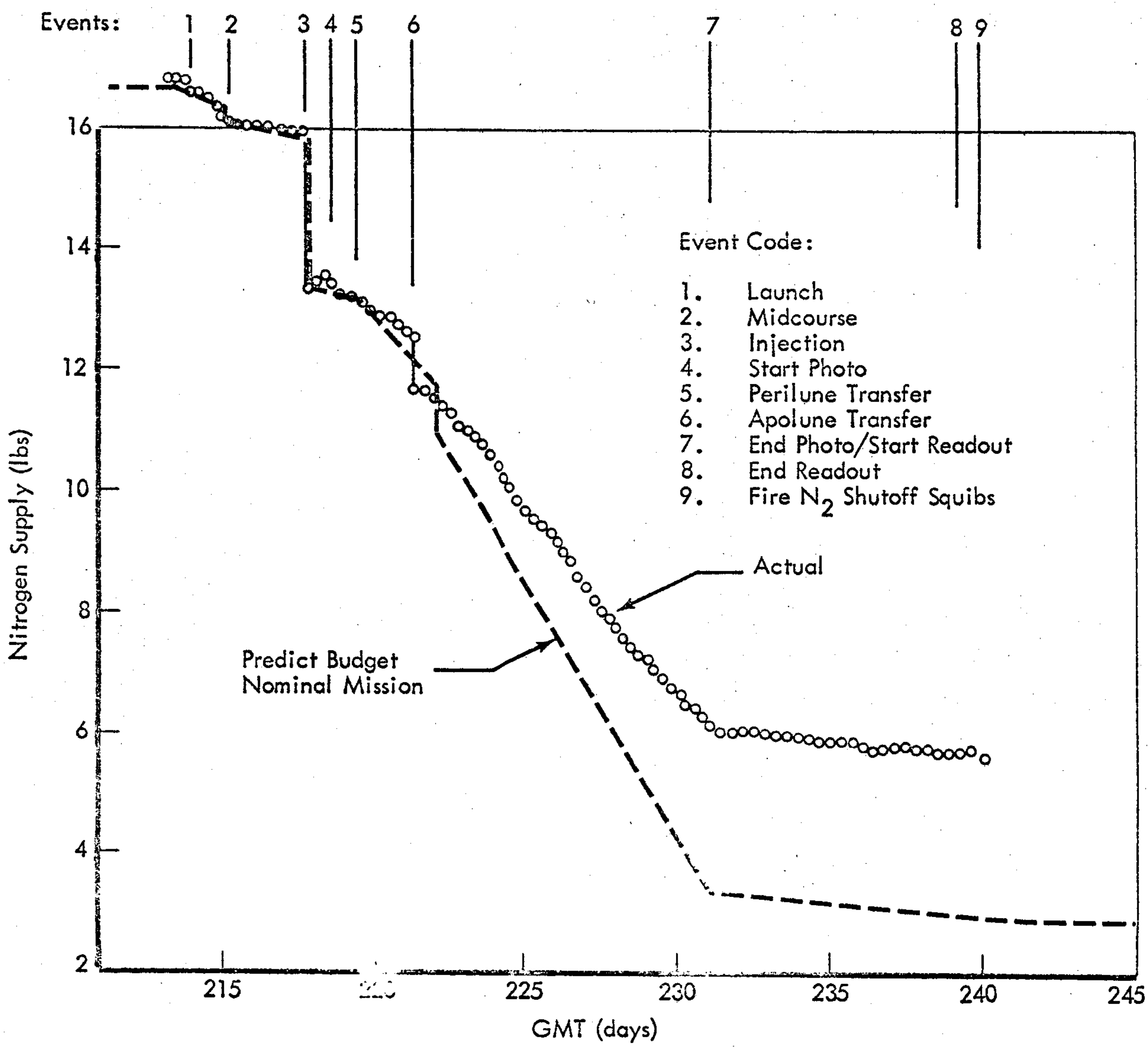


Figure 3-4: Velocity Control Subsystem Available Nitrogen History — Spacecraft Flight Data

- A greater-than-actual allowance was made for "acquire Sun" maneuvers. The gyro drift rates were minimal to the degree that Sun acquisition was essentially just a pitch maneuver (from an off-Sun attitude) rather than a pitch and yaw maneuver.
- Nitrogen consumption for attitude maneuvers was calculated on the basis of a heavier spacecraft than was actually the case.

All factors combined to reduce the nitrogen consumption rate during final-orbit photography to an overall average of 0.584 pound per day rather than the predicted value of approximately 0.85 pound per day. During 8 days of final readout, the average consumption rate was found to be 0.053 pound per day as compared to a predicted value of 0.05 pound per day.

Figure 3-5 shows the variations in subsystem pressures during flight. The lack of Sun occultation periods produced the essentially constant propellant tank pressure trend, the slight increase matching the slow temperature rise shown in Figure 3-6. No system leakage was apparent.

Figure 3-6 plots subsystem temperature trends in a similar manner (though at only 12-hour intervals). As noted, local temperatures were generally in the region of 50 to 75°F; the transients in engine valve temperature (AT03) are the result of propulsive maneuvers. The temperature characteristics are similar to those observed during Mission IV.

Maneuver Performance -- During the primary photographic mission of Lunar Orbiter V, the velocity control subsystem provided four propulsive maneuvers. These consisted of a midcourse maneuver, an initial orbit injection maneuver, a perilune transfer maneuver to place the spacecraft in an interim orbit, and an apolune transfer maneuver to achieve the final orbit for high-resolution photography (initial and interim orbits were used for survey photography). The subsystem performance summary is presented in Table 3-2. The maneuvers indicate that the system had a delivered thrust of approximately 100 pounds at a specific impulse of 276 seconds. For comparison, the engine on Spacecraft demonstrated the following performance characteristics during the four 5-second steady-state acceptance test runs:

Thrust, lbs	98.6
Specific impulse, sec	277.7
Mixture ratio, o/f	2.013

Engine acceptance data are normalized to a standard propellant temperature of 70°F; an average value of propellant temperature (ST04) at the time of the maneuvers was on the order of 50 to 52°F.

Figures 3-7 and 3-8 present VCS telemetry data obtained during the orbit injection maneuver; Figure 3-7 shows pressure and temperature data, while Figure 3-8 plots dynamic data in the form of gimbal actuator positions and accelerometer output. Gimbal actuator movement was minimal as on previous flights, and the characteristics of velocity change were nominal.

Engine valve temperature during and following each maneuver was normal. A brief summation of maximum valve temperature (AT03), resulting from thermal soak-back, is presented in Table 3-4.

Table 3-4: Engine Valve Temperature Maximum Soak-Back

Midcourse	100.1°F
Injection	109.8
Perilune transfer	94.8
Apolune transfer	107.7

The gimbal actuator movements presented in Figure 3-7 are typical of this, and previous, missions. Table 3-5 summarizes actuator positions before and after each maneuver.

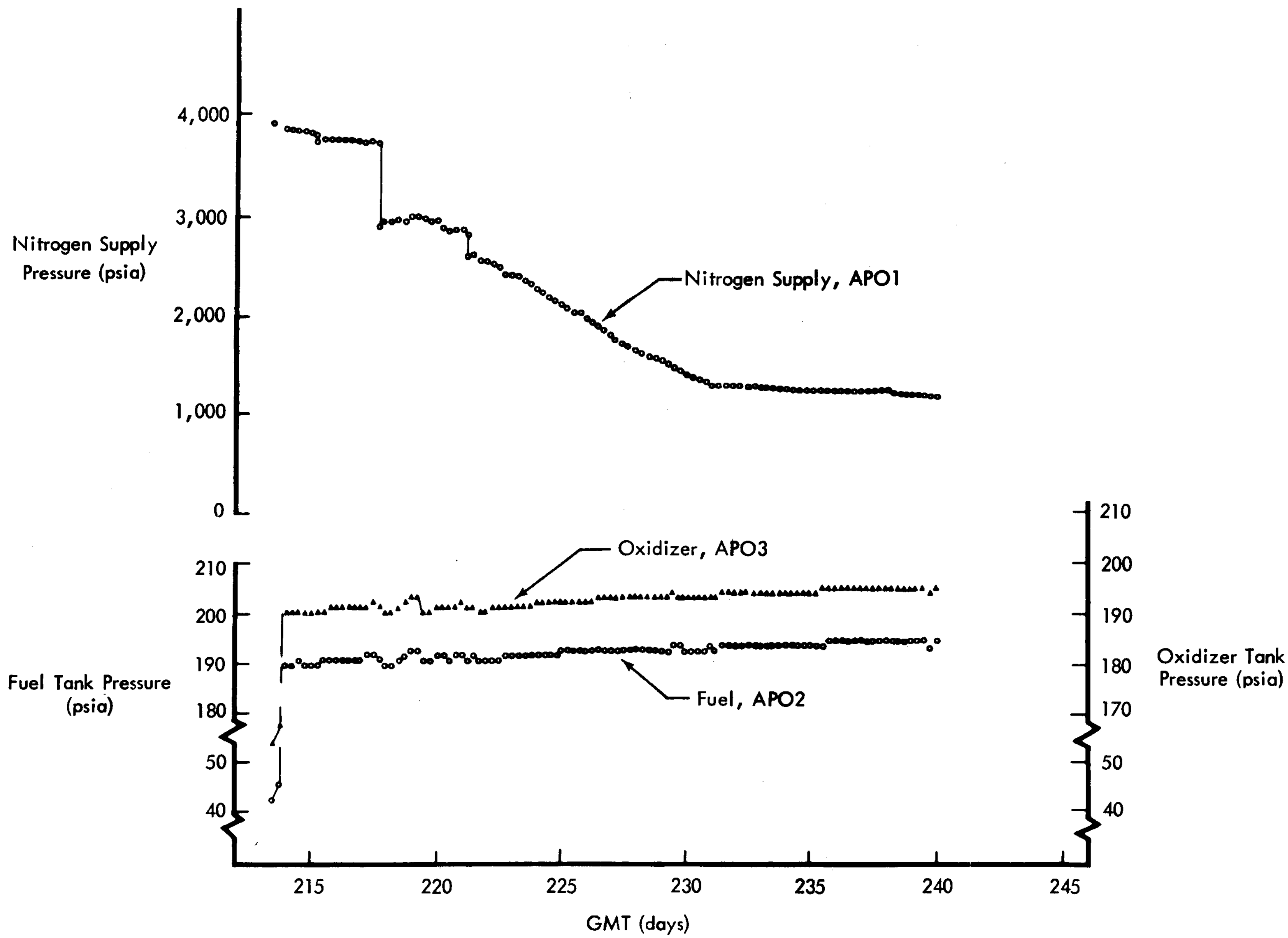


Figure 3-5: Velocity Control Subsystem Pressure-Time Histories – Spacecraft Flight Data

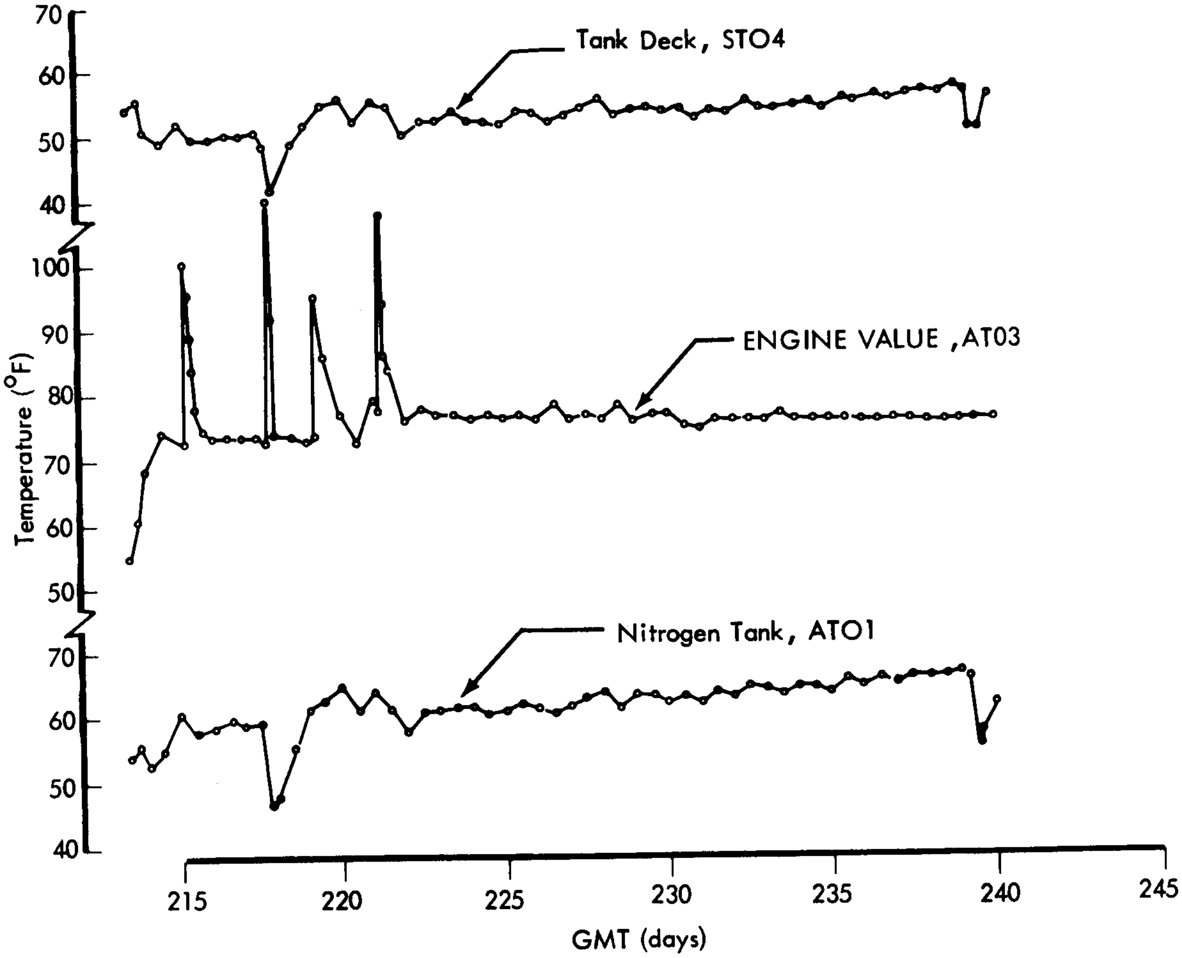
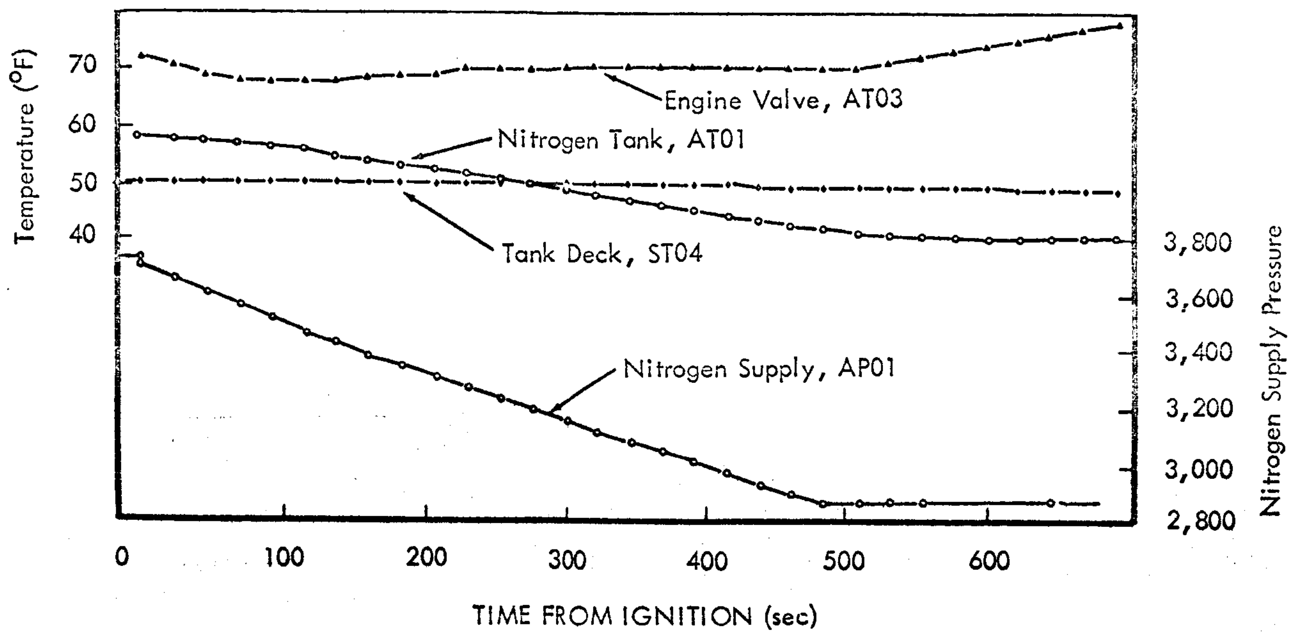


Figure 3-6: Velocity Control Subsystem Temperature-Time Histories – Spacecraft Flight Data

Table 3-5: Gimbal Actuator Position

	Pitch (deg)		Yaw (deg)	
	Pre-	Post-	Pre-	Post-
Launch	-0.062	-0.062	0.040	0.040
Midcourse	-0.039	-0.062	0.063	0.496
Injection	-0.085	-0.085	0.473	0.314
Perilune transfer	-0.085	-0.039	0.336	0.268
Apolune transfer	-0.039	-0.131	0.268	0.473

Slight discrepancies between the conclusion of one maneuver and the beginning of the next are reflections of data resolution characteristics.



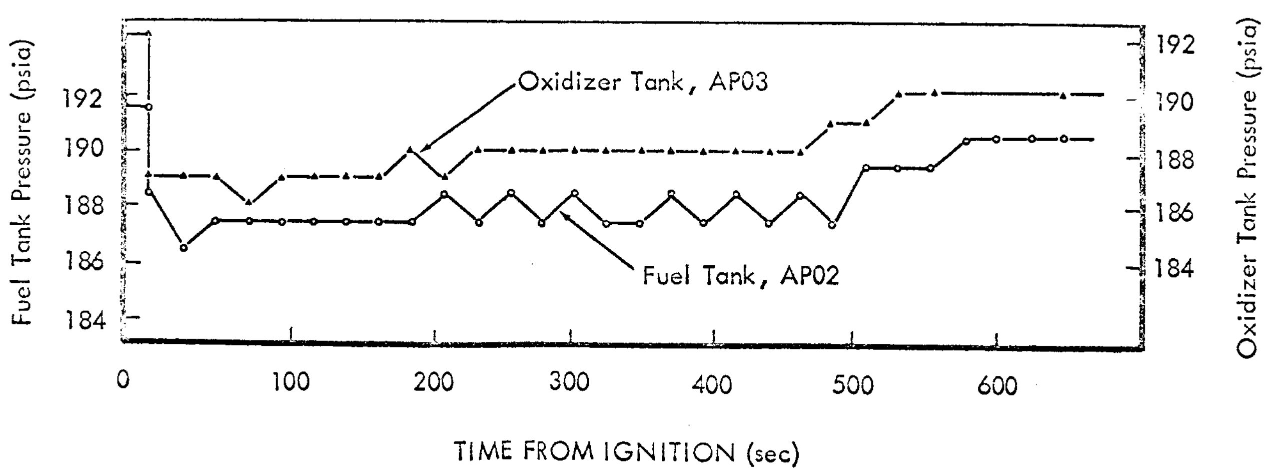


Figure 3-7: Velocity Control Subsystem Orbit Injection Maneuver System Pressures and Temperatures

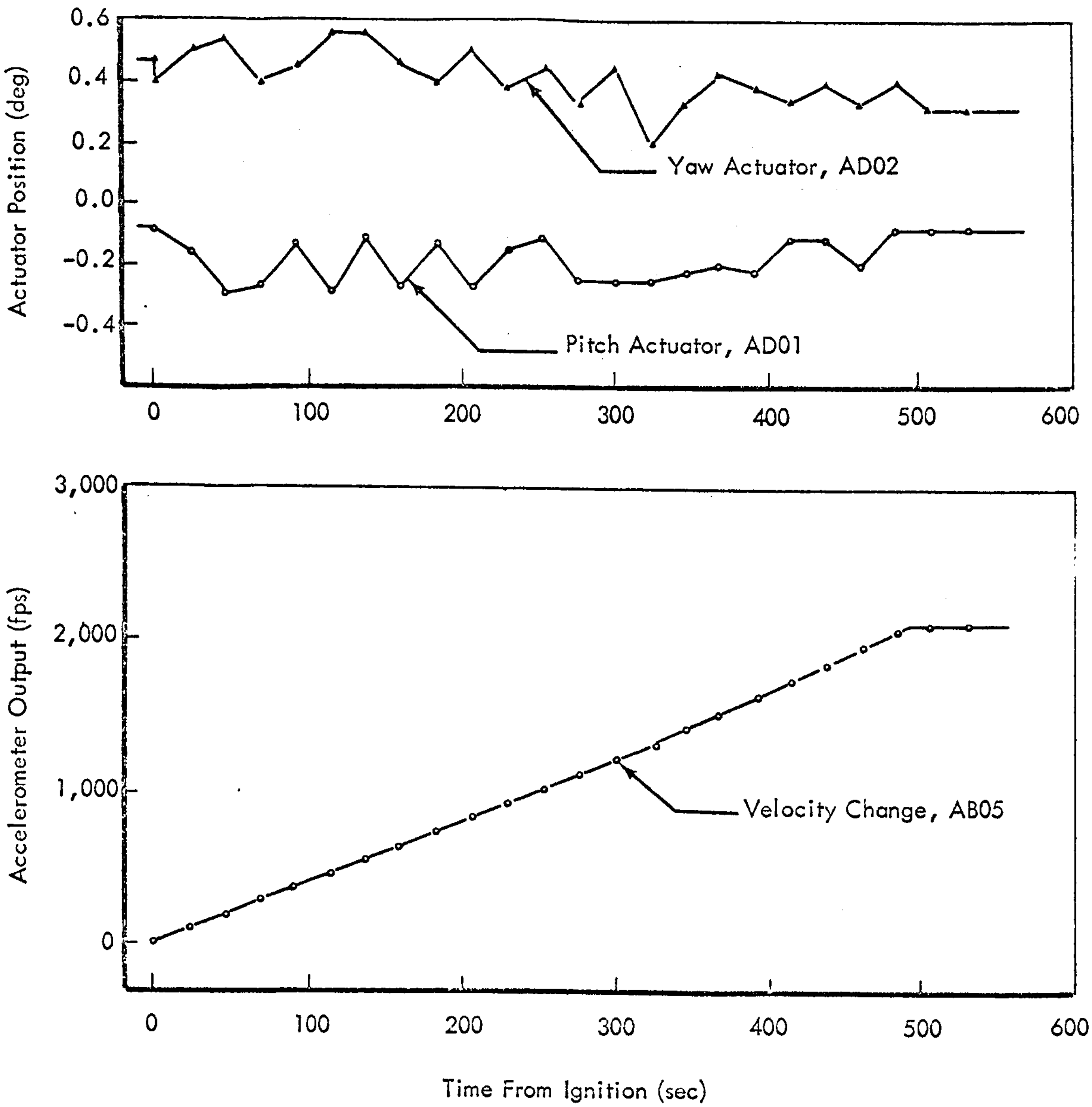


Figure 3-8: Velocity Control Subsystem Orbit Injection Maneuver System Dynamics

3.6 STRUCTURES AND MECHANISMS SUBSYSTEM

Presented here is a brief discussion of factors relating to the structures and mechanisms of Lunar Orbiter V. This involves a presentation of vibration data observed during launch, deployment and squib actuation sequencing, and camera thermal door operational history. One micrometeoroid impact was observed during the primary mission.

3.6.1 Launch Vibration Environment

Figures 3-9 through 3-26 present vibration data (as recorded from Agena telemetry) from liftoff Day 213, 22:33:00.338 GMT to Agena second cutoff at 23:05:48.0 GMT. For comparison, the upper envelope of spacecraft vibration testing is included. These data are comparable to that observed during previous flights.

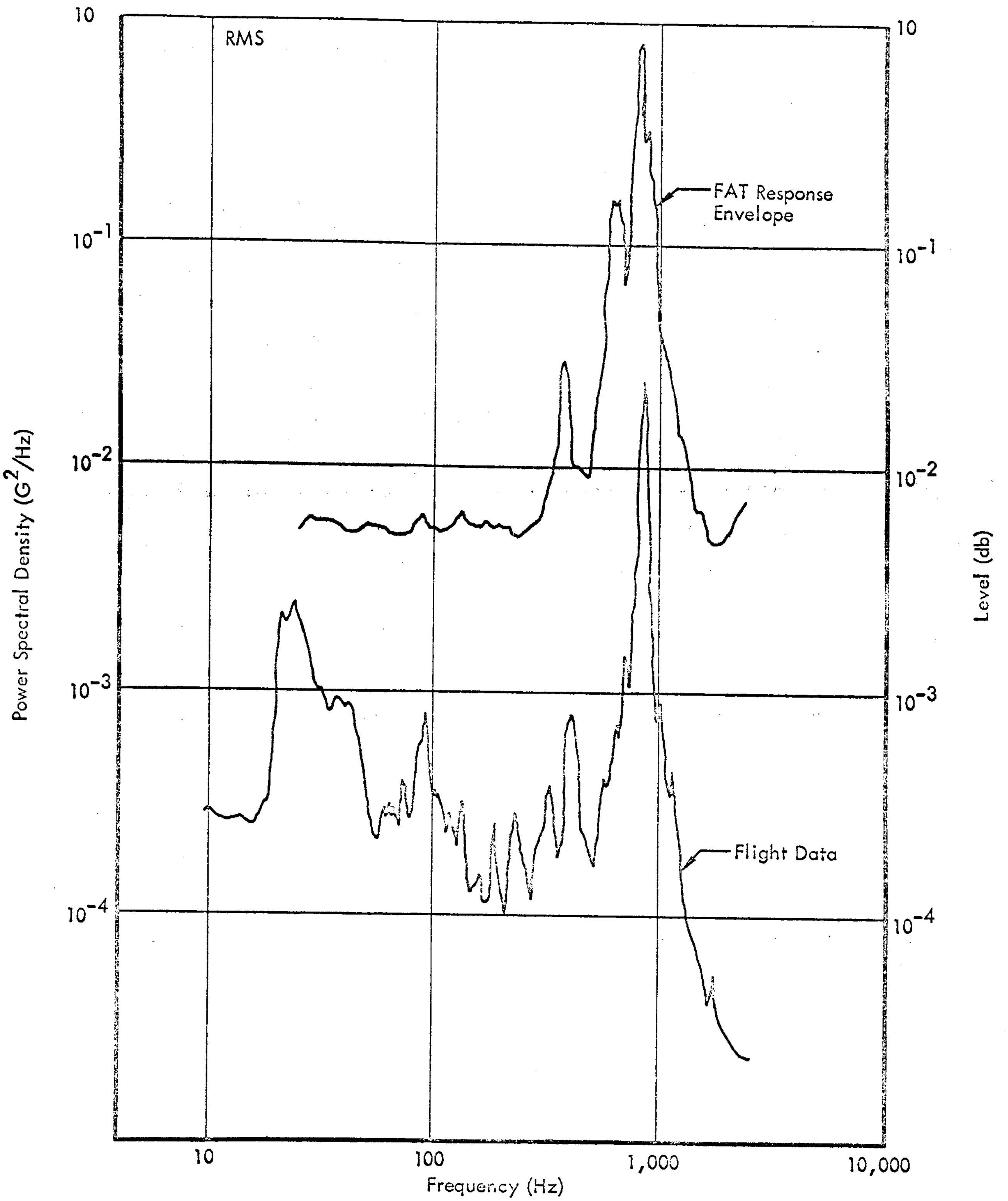


Figure 3-9: Random Response – Liftoff – Longitudinal Accelerometer (214x) (T + 0 Sec. To T + 3 Sec.)

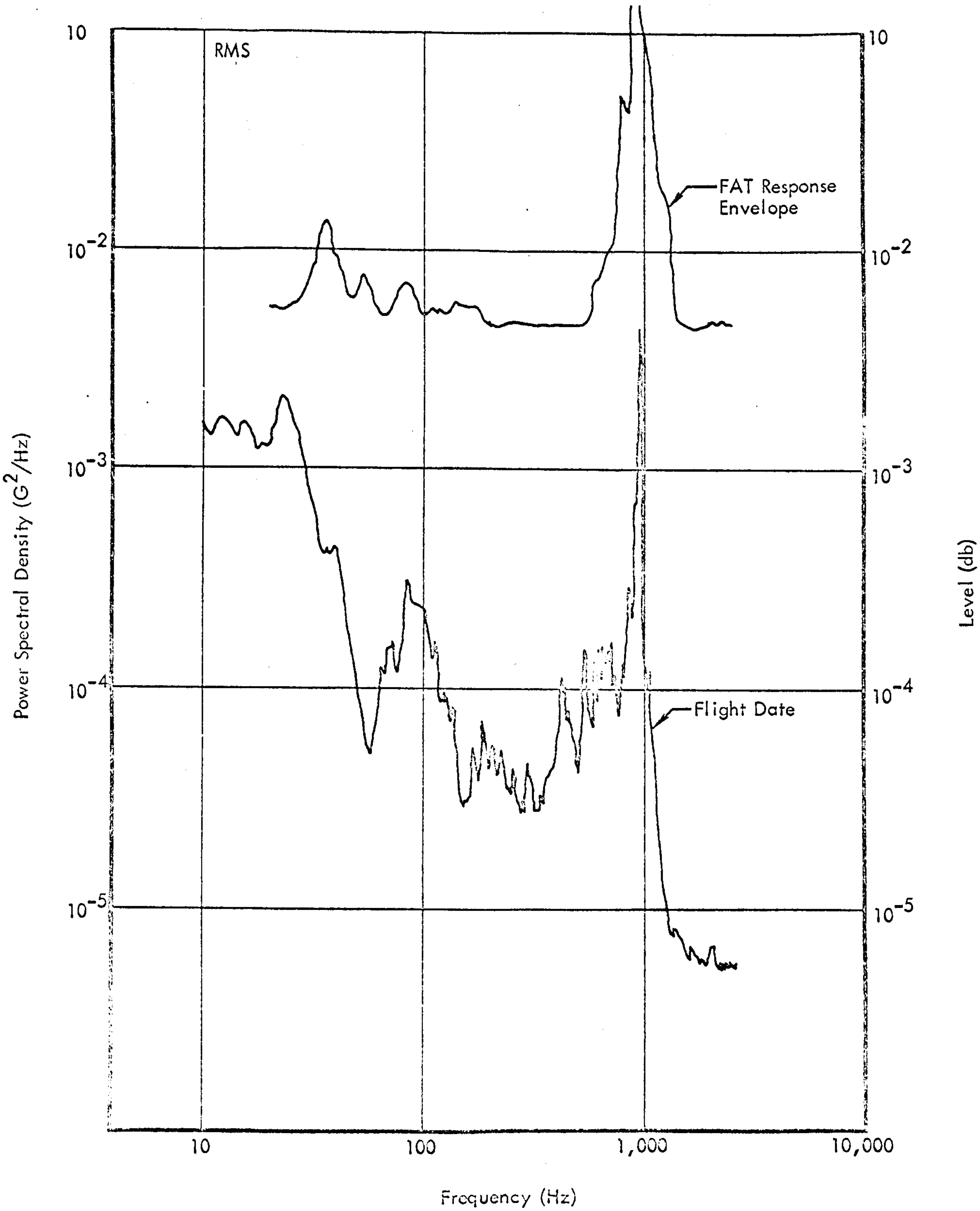


Figure 3-10: Random Response – Liftoff – Lateral Accelerometer (210z)

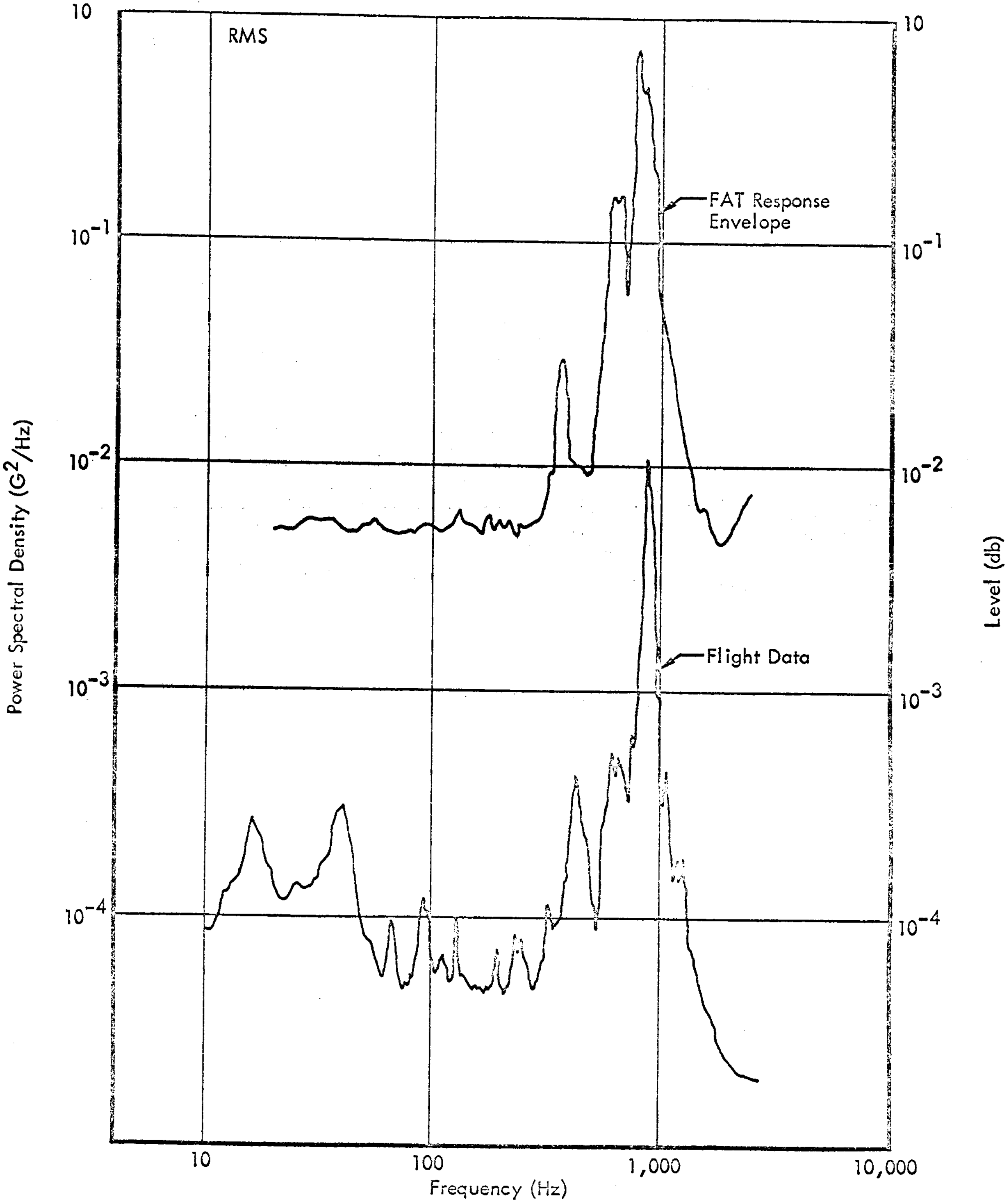


Figure 3-11: Random Response — Transonic Longitudinal Accelerometer (214x) (T + 49 Sec. To T + 59 Sec.)

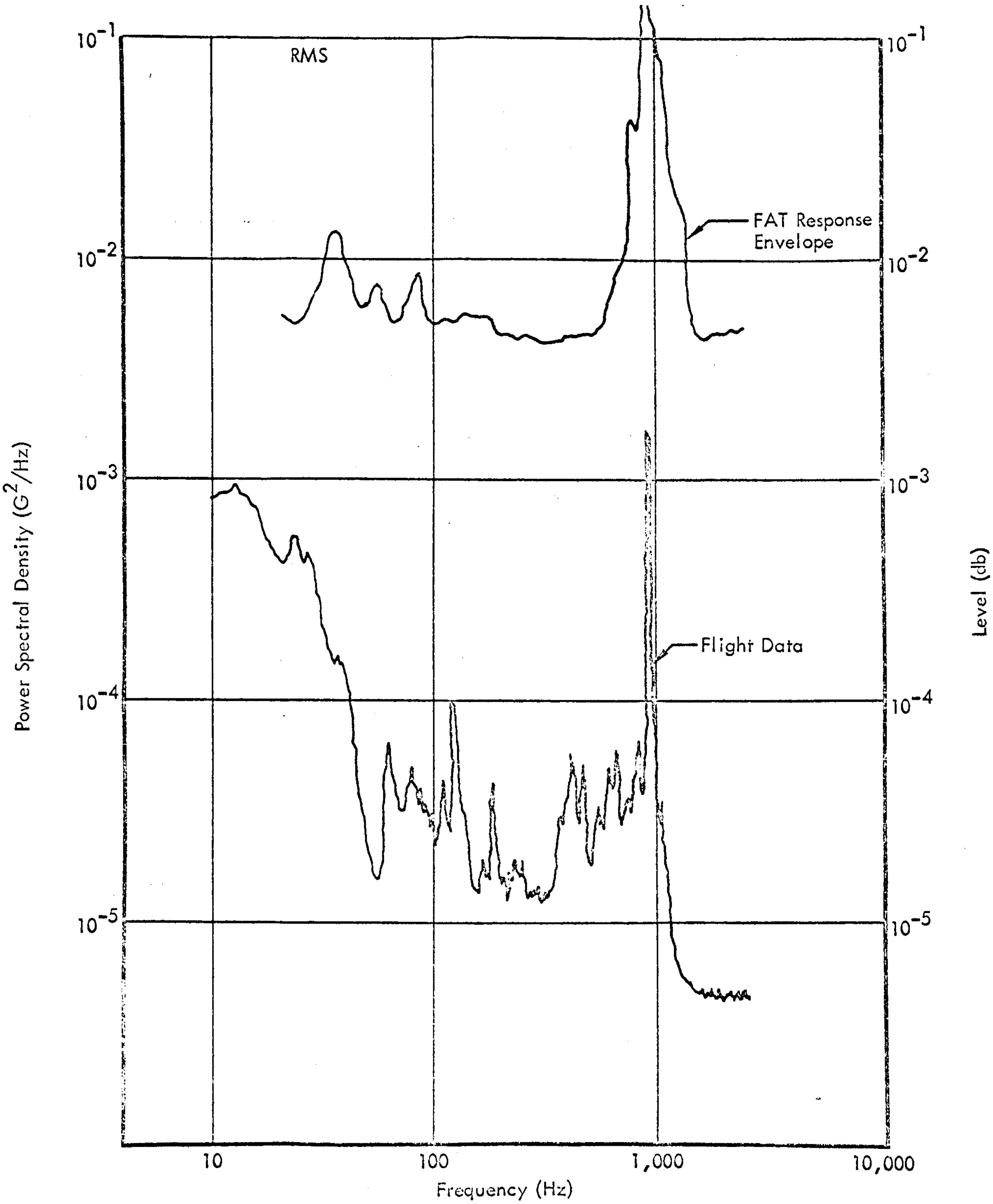


Figure 3-12: Random Response — Transonic Lateral Accelerometer (210 z)
(T + 49 Sec. To T + 59 Sec.)

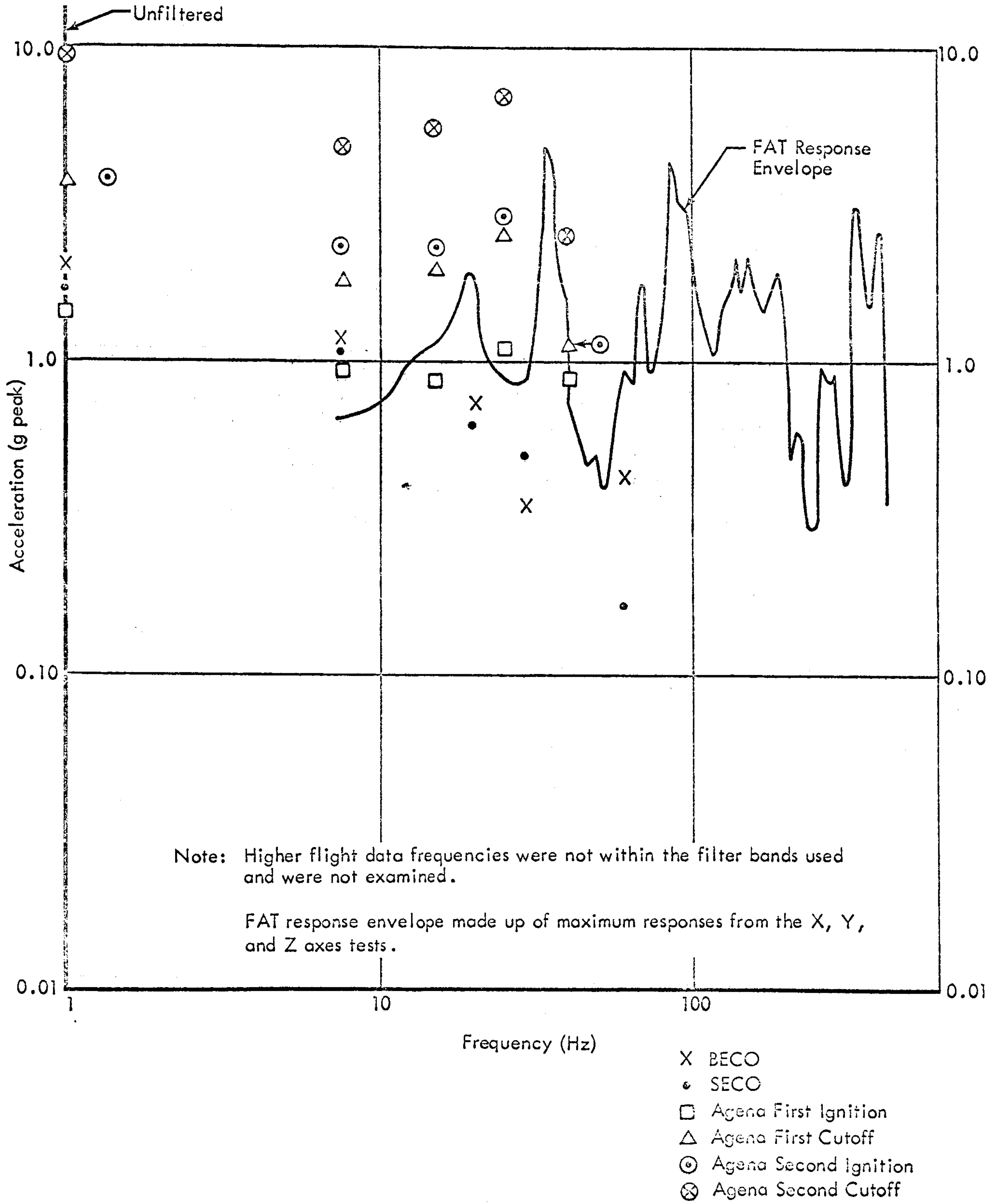


Figure 3-13: Spacecraft Vibration – Peak Longitudinal Response (214x) – Flight Data vs FAT Response Envelope

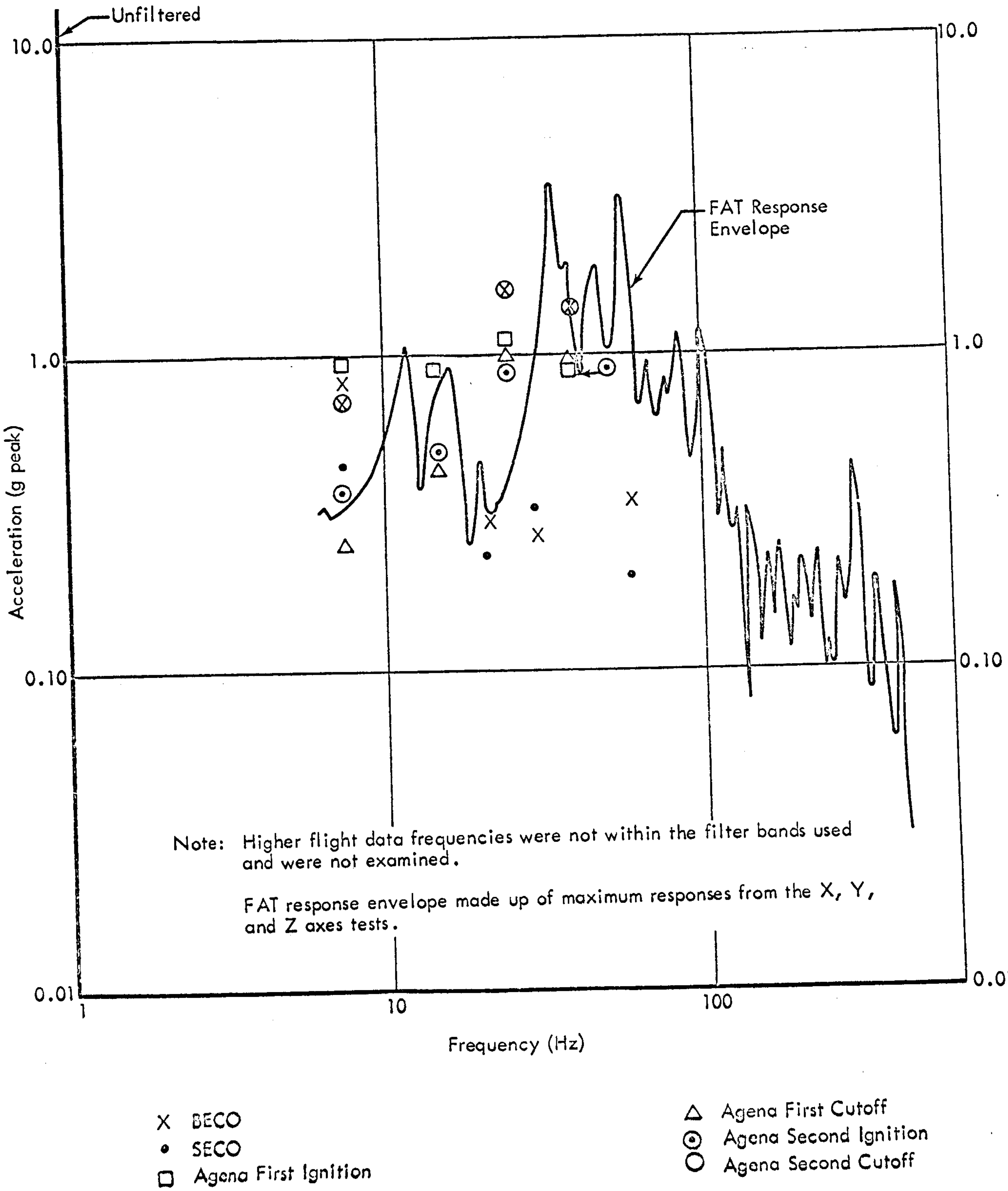


Figure 3-14: Spacecraft Vibration – Peak Lateral Response (210z) – Flight Data vs FAT Response Envelope

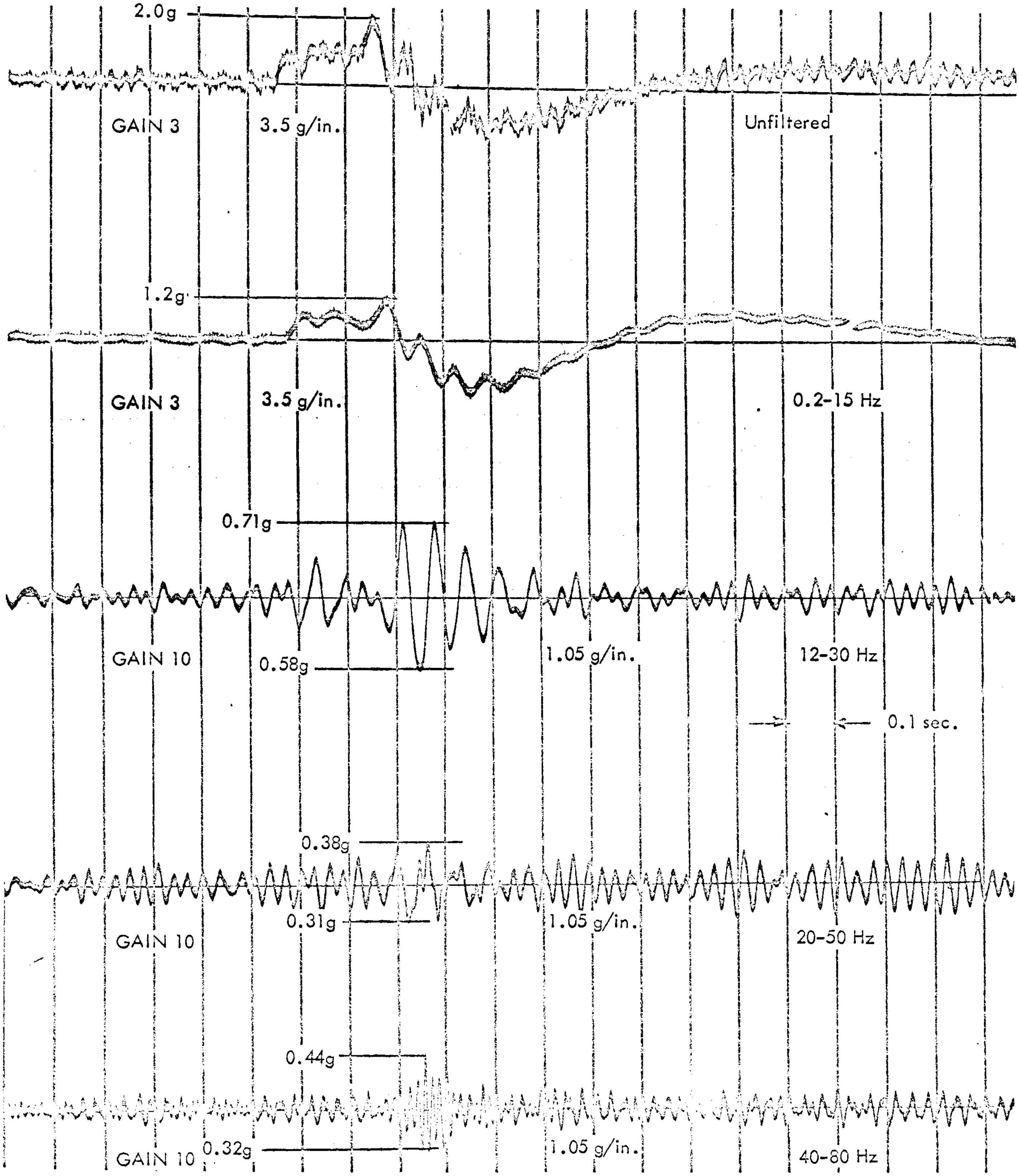


Figure 3-15. BECO Longitudinal Vibration (214x)

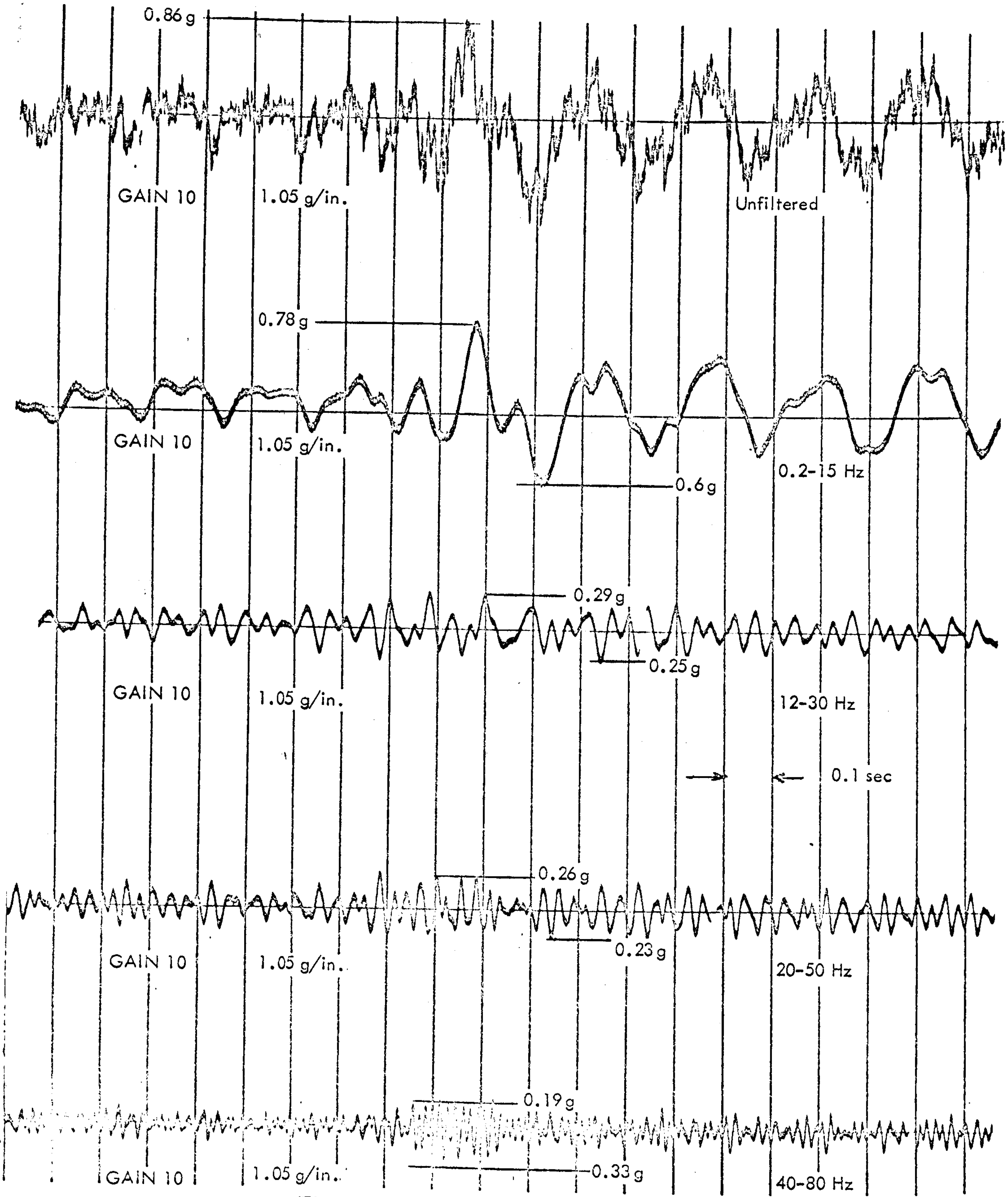


Figure 3-16: BECO Lateral Vibration (210z)

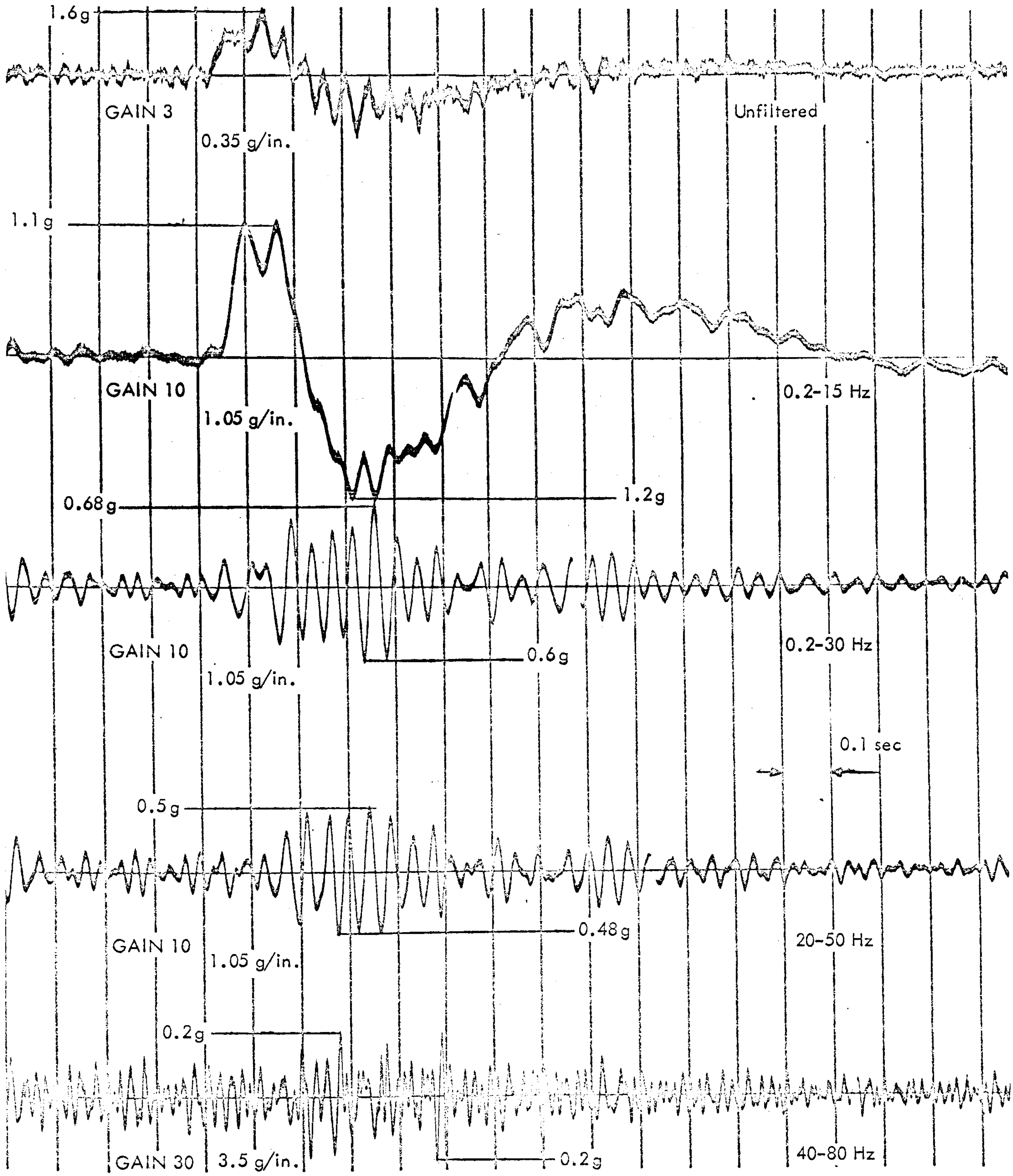


Figure 3-17: SECO Longitudinal Vibration (214x)

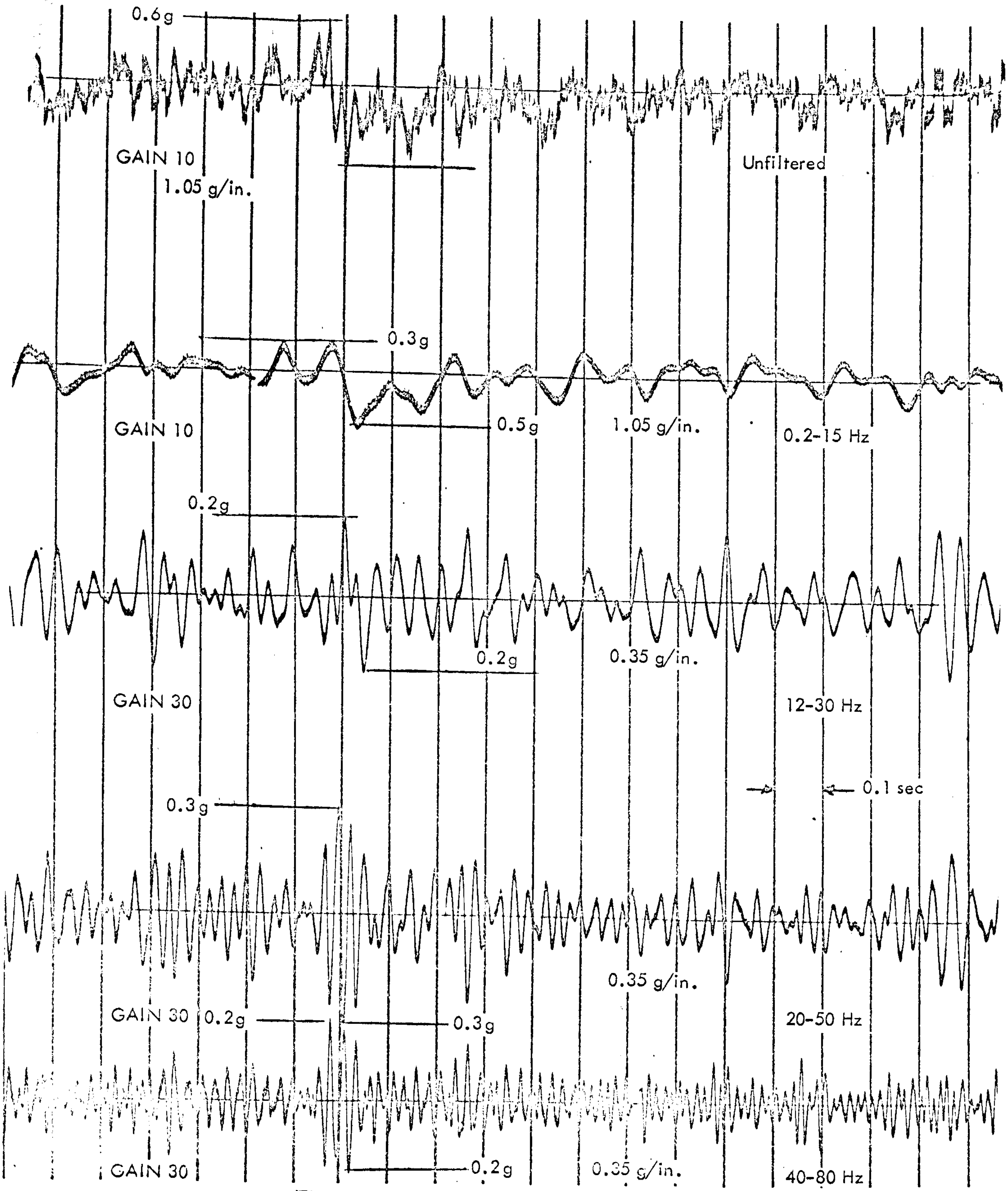


Figure 3-18: SECO Lateral Vibration (210z)

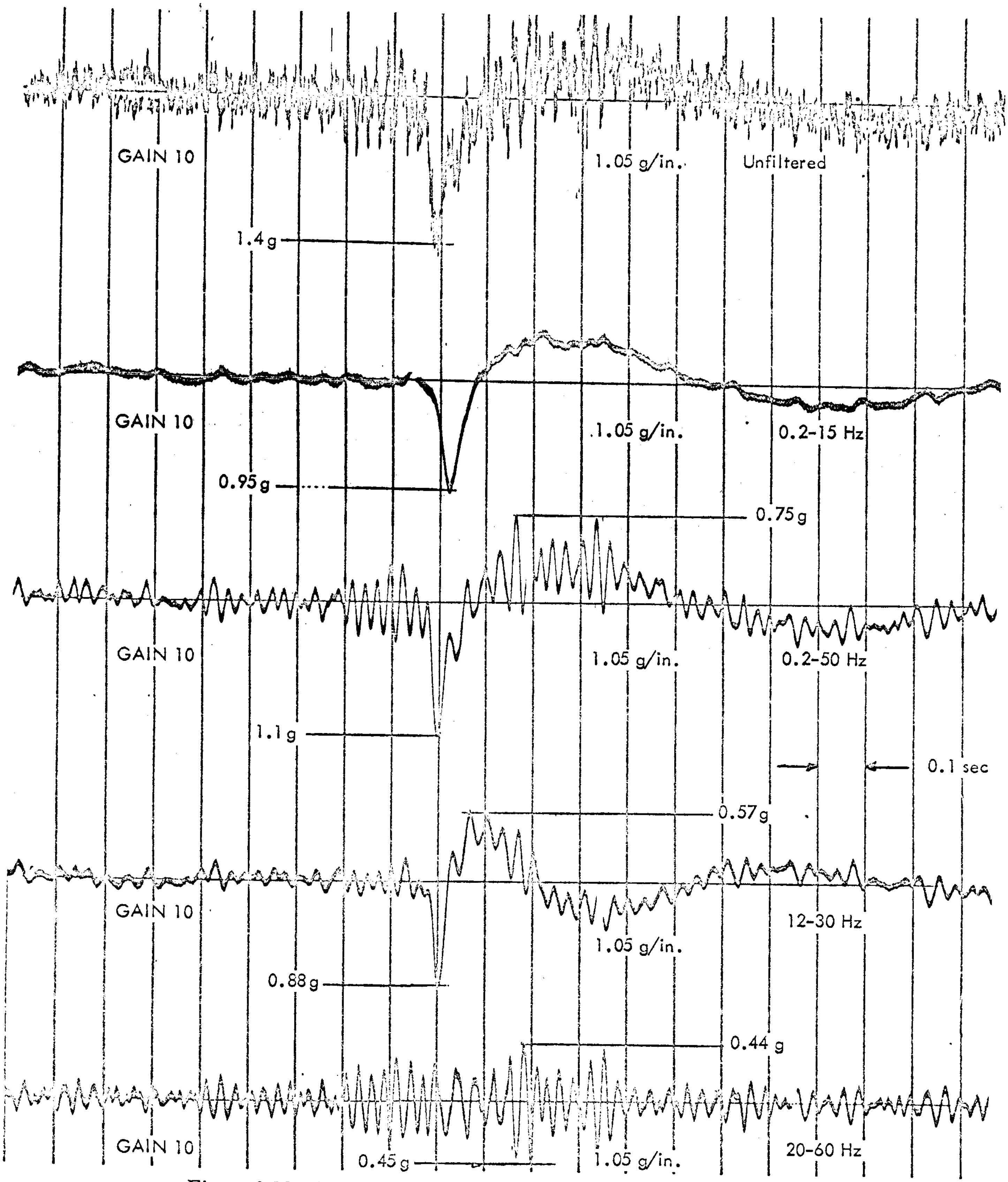


Figure 3-19: Agena First Ignition — Longitudinal Vibration (214x)

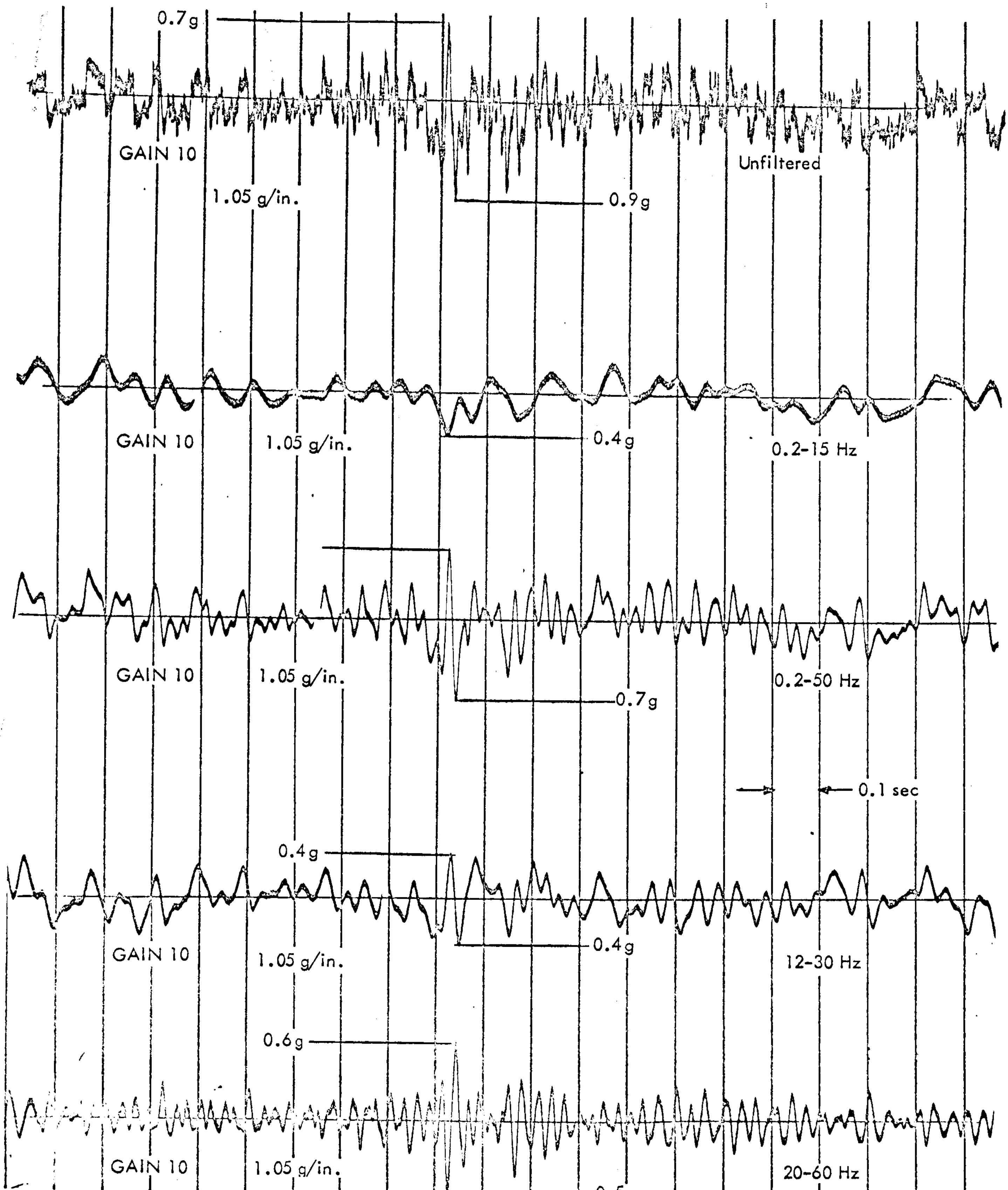


Figure 3-20: Agena First Ignition – Lateral Vibration (210z)

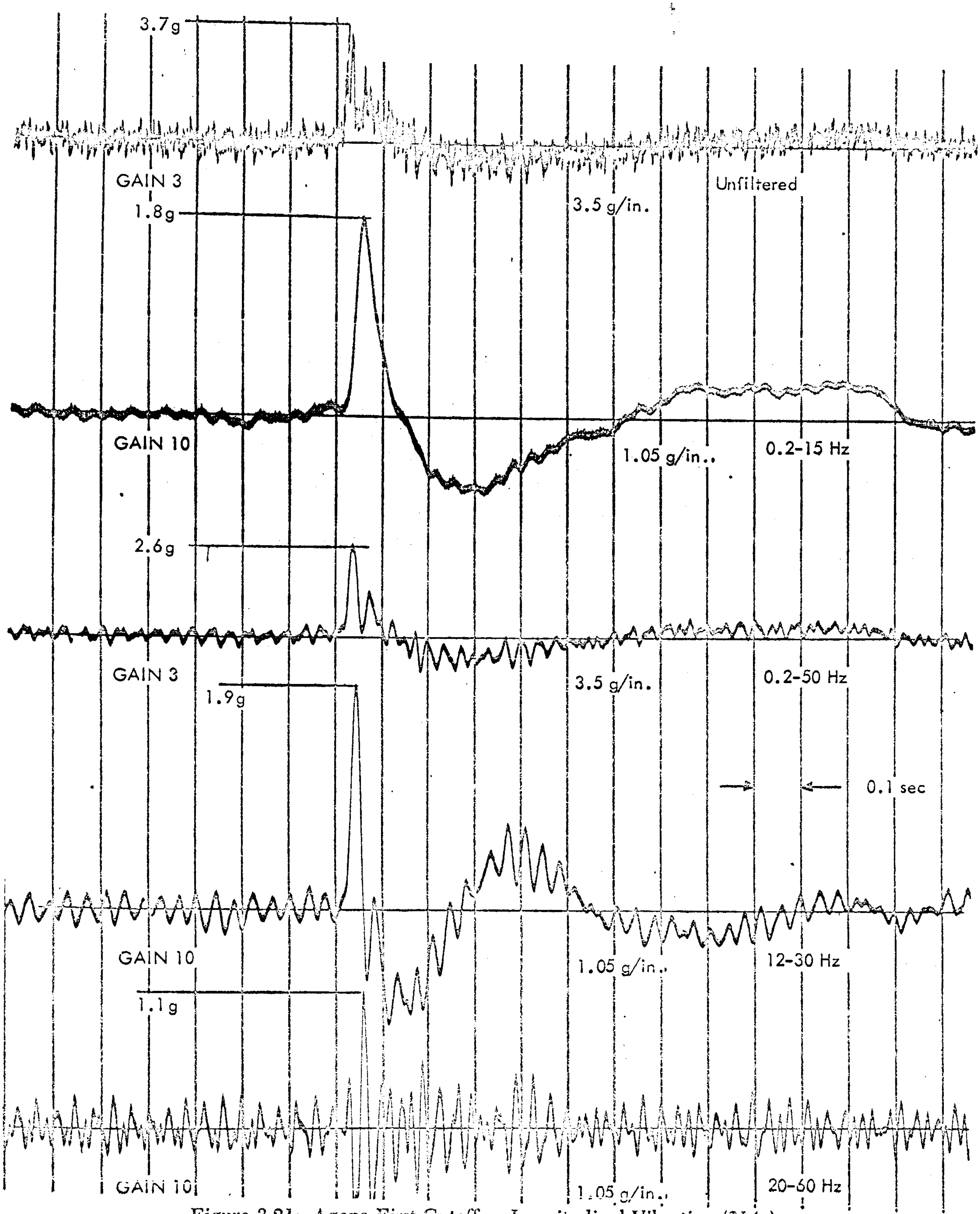


Figure 3-21: Agena First Cutoff – Longitudinal Vibration (214x)

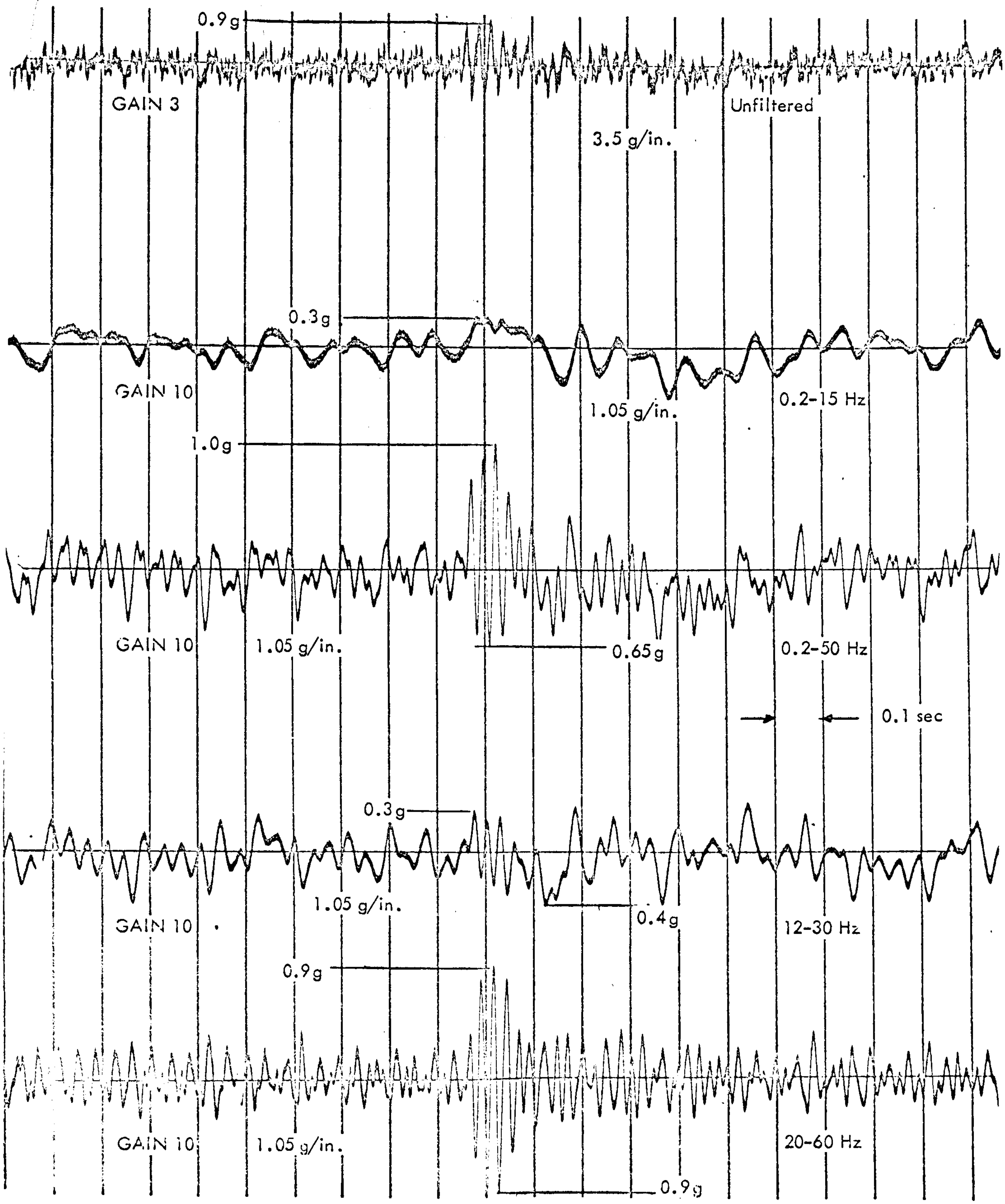


Figure 3-22: Agena First Cutoff – Lateral Vibration (2104)

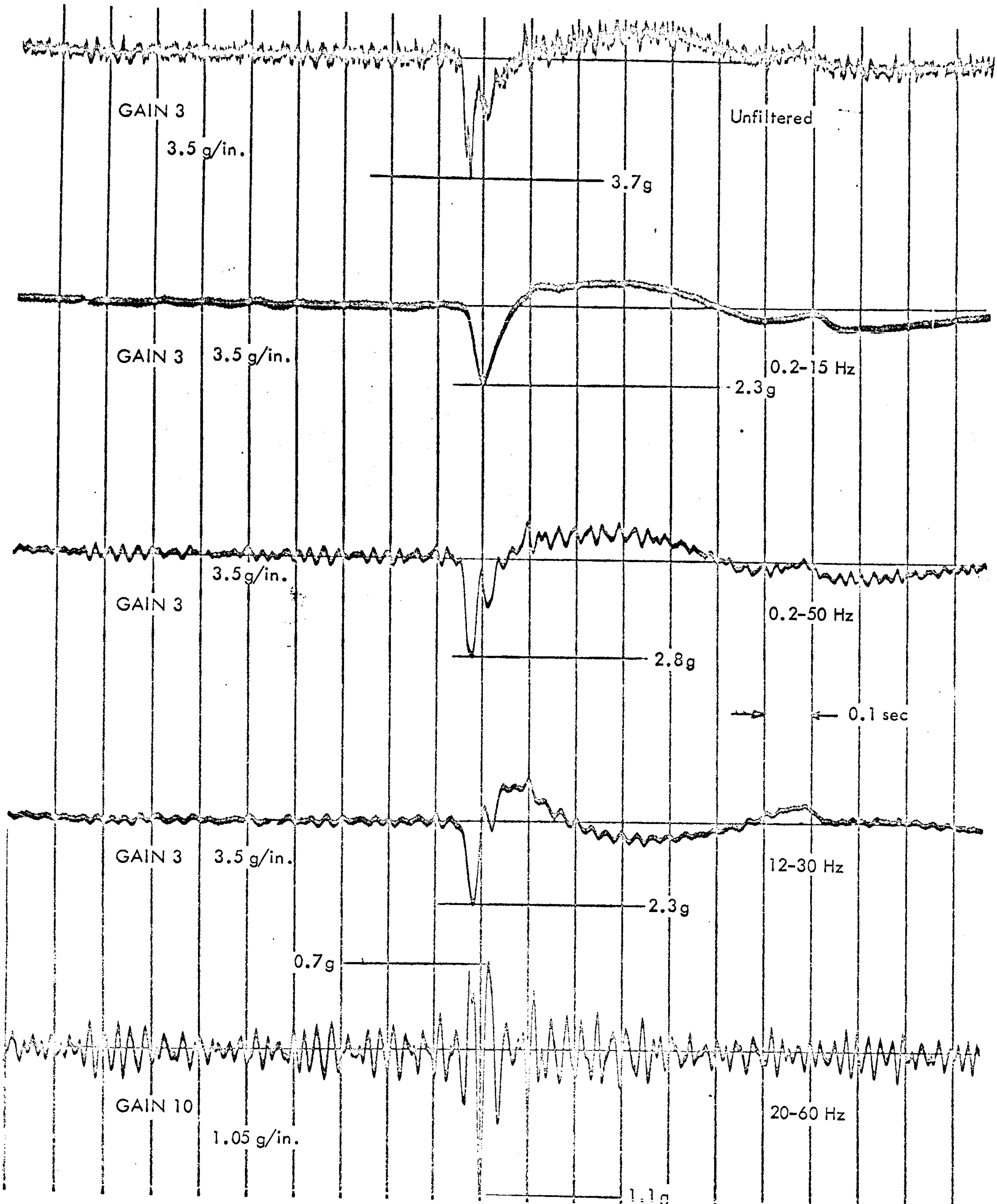


Figure 3-23: Agena Second Ignition – Longitudinal Vibration (214x)

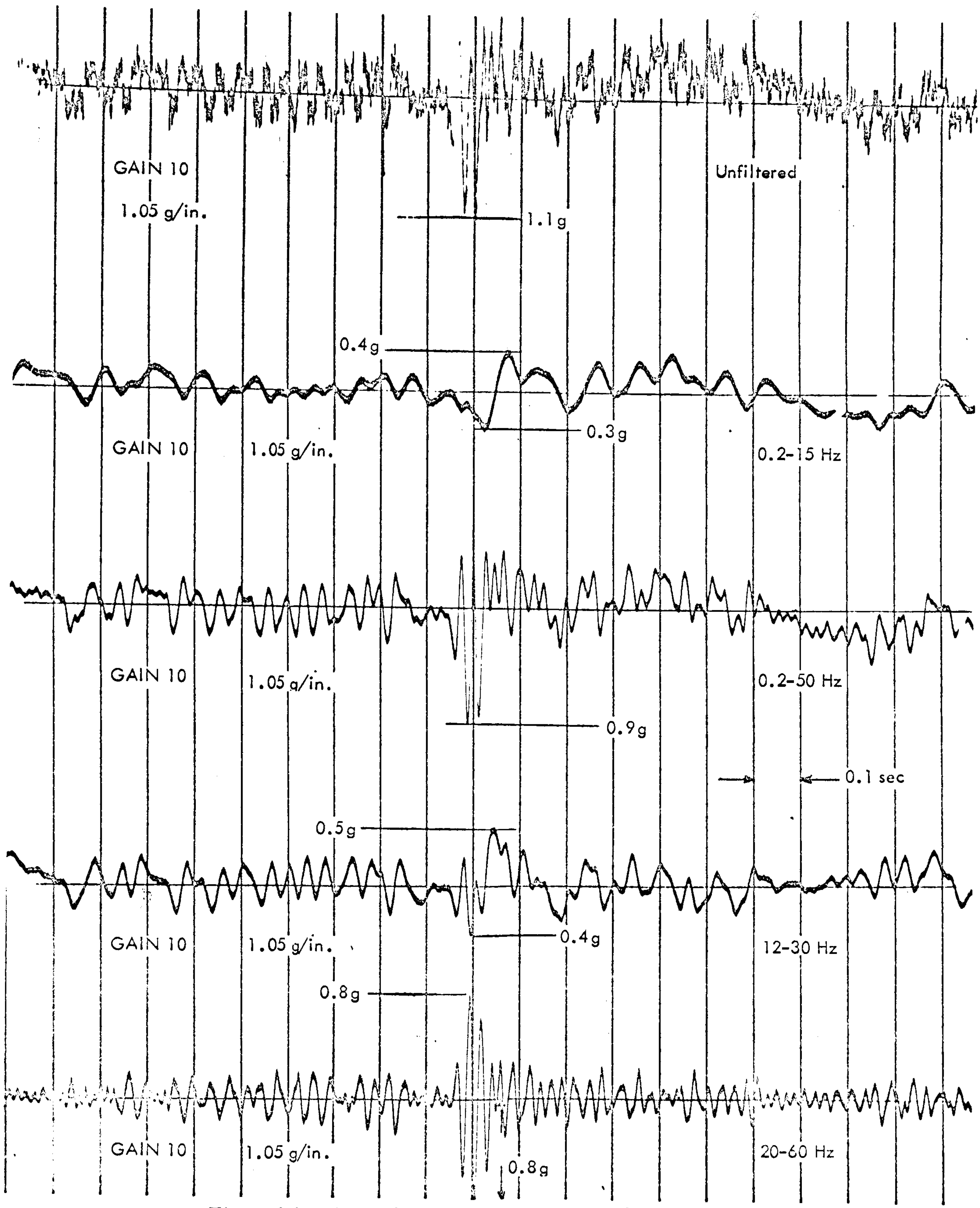


Figure 3-24: Agena Second Ignition – Lateral Vibration (210z)

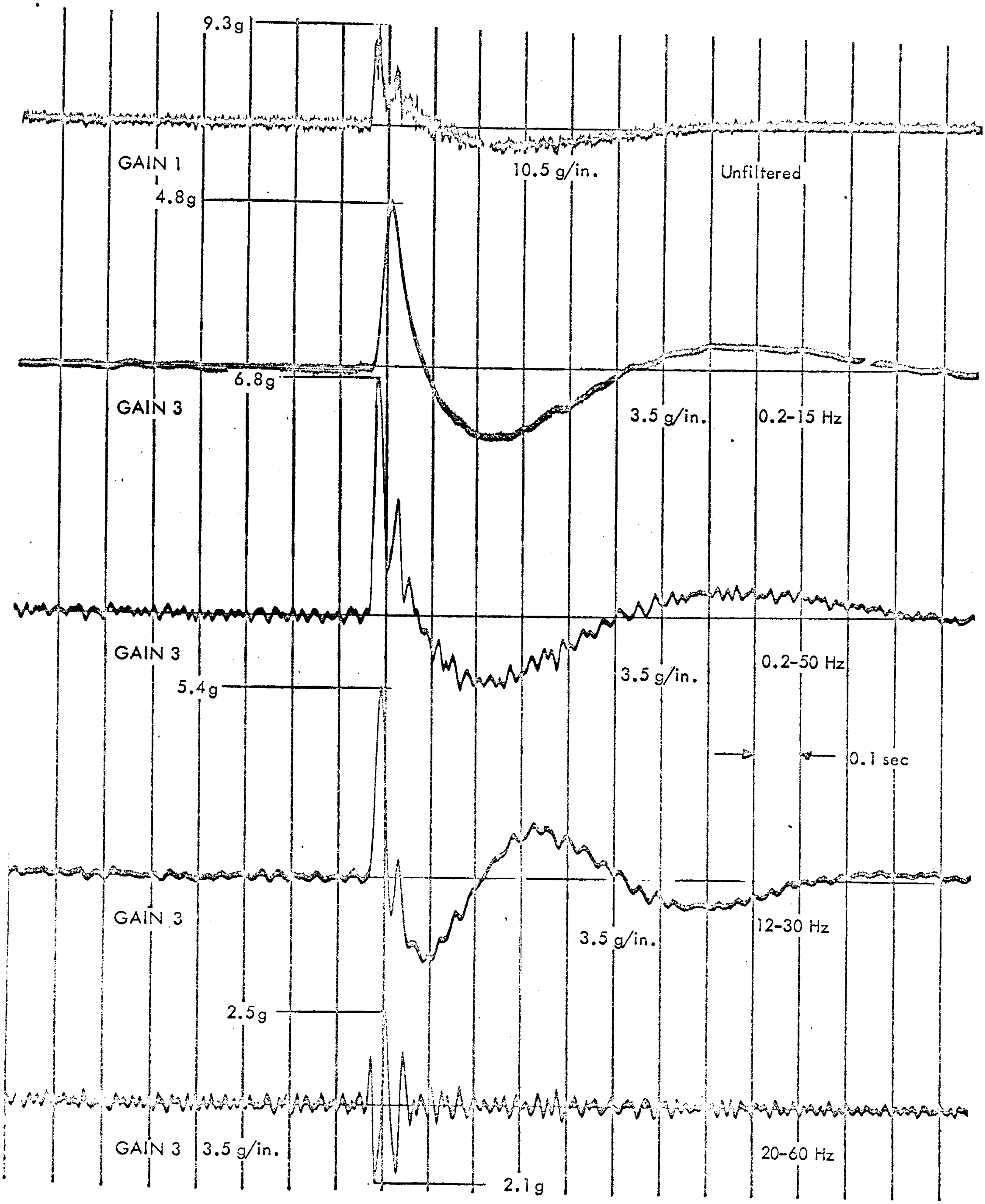


Figure 3-25: Agena Second Cutoff – Longitudinal Vibration (214x)

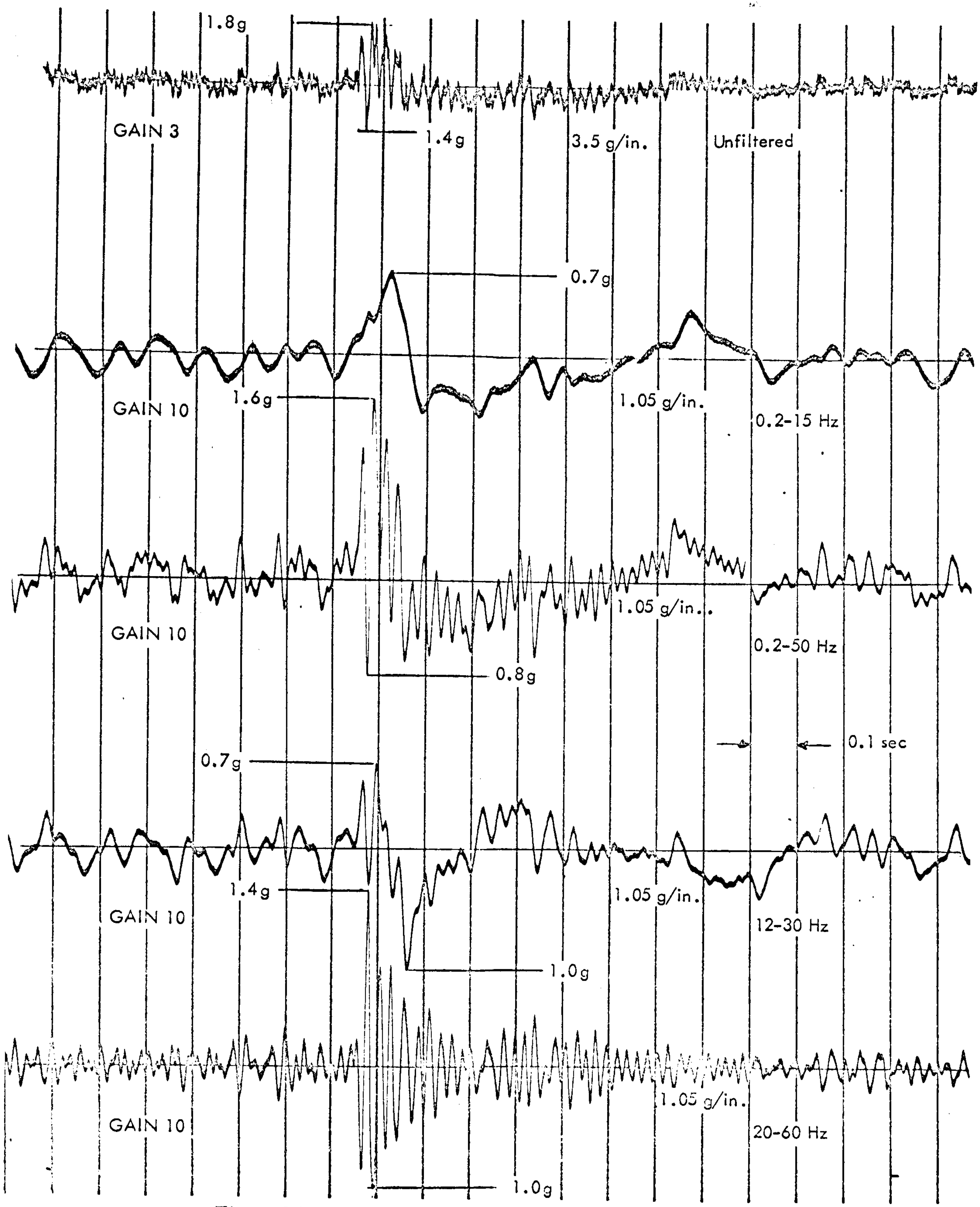


Figure 3-26: Agena Second Cutoff – Lateral Vibration (210z)

3.6.2 Deployment and Squib Actuation

Following spacecraft separation from the Agena at 23:08:33.8 GMT, the deployment sequence was initiated. Based on the stored program commands, antenna deployment was initiated at 23:10:32.0 GMT, solar panel deployment commenced at 23:10:57.8 GMT, and the nitrogen isolation squib valve was actuated at 23:11:49.4 GMT. Upon receipt of first good data from DSS-41 at 23:22 GMT, it was verified that all events had been successfully accomplished. The VCS propellant isolation squib valves were successfully actuated on Day 214, at 14:28:28 GMT. Just prior to the conclusion of the primary mission, the nitrogen shutoff squib valve was actuated at 22:38 GMT on Day 239.

3.6.3 Camera Thermal Door

During Mission V, the camera thermal door operated without incident. On Day 215, during the cislunar phase, the unit was cycled three times as part of a special test. For lunar photography, the camera door was cycled on 74 occasions; the overall mission total was 77 cycles.

3.6.4 Thermal Control

The thermal control subsystem of the Lunar Orbiter spacecraft is a passive system with the equipment mounted on a Sun-oriented equipment mounting deck (EMD). Heat generated by equipment is conducted to the EMD where it is radiated to the space environment. The EMD is covered with paint (B1056 with an overcoat of S13G) having a low solar absorptance. A uniform, 20% distribution of mirrors (optical solar reflectors) is bonded to the paint on the EMD, with an additional 10% concentration of mirrors placed in sections of the EMD where additional heat rejection is needed. Thermal control is achieved by varying the attitude of the EMD with respect to the Sun. The equipment is enclosed in multilayer blanket insulation, and supplemental heating is supplied, as needed, to the propellant and photo subsystems by electric heaters.

Spacecraft temperatures were maintained within prescribed temperature limits throughout the mission. In general, Mission V was the most successful of any mission and was devoid of thermal anomalies.

Equipment mount deck thermal coating (paint) degradation occurred, similar to that experienced with Spacecraft 7. However, degradation of the thermal coating caused no impairment of the spacecraft mission objectives, because its effect on the spacecraft was offset by pitching the spacecraft off the sunline at a predetermined angle. These maneuvers maintained temperatures at the desired levels.

3.6.4.1 Thermal Problems

The only thermal problem experienced with Spacecraft 3 was thermal paint degradation similar to Mission IV.

Figure 3-27 compares the absorptivity of the thermal control coating of Spacecraft 3 with that of Spacecraft 5. The effective absorptivity of the paint-mirror combination on Spacecraft 5 is compared to the absorptivity of the paint alone on Spacecraft 3.

The low altitude of L/O V makes the EMD subject to heat radiated from Moon and sunlight reflected from the Moon. This fact, along with the photo maneuvers, made the orbit data valueless in computing degradation data. The only usable data was obtained late in the mission, when the spacecraft was put on Sun for approximately 4 hours in Orbits 150 through 151.

The degradation rates were sufficiently low so that EMD temperatures were not a major problem during the mission. The spacecraft was pitched off the Sun during readout to reduce the EMD and component temperatures, particularly the EMD under the camera and TWTA temperature. The maximum off-Sun angle used was 41 degrees.

3.6.4.2 Thermal Paint Experiment

An experiment involving four types of thermal paints was included onboard the Mission V Lunar Orbiter spacecraft. The purpose of this experiment was to evaluate the stability of these paints when exposed to solar radiation for long periods in a deep space environment. The paint samples included in this experiment were S13G overcoat on B1056, Hughes organic white paint, McDonnell Z-93 white paint, and a silicone overcoat on aluminum foil.

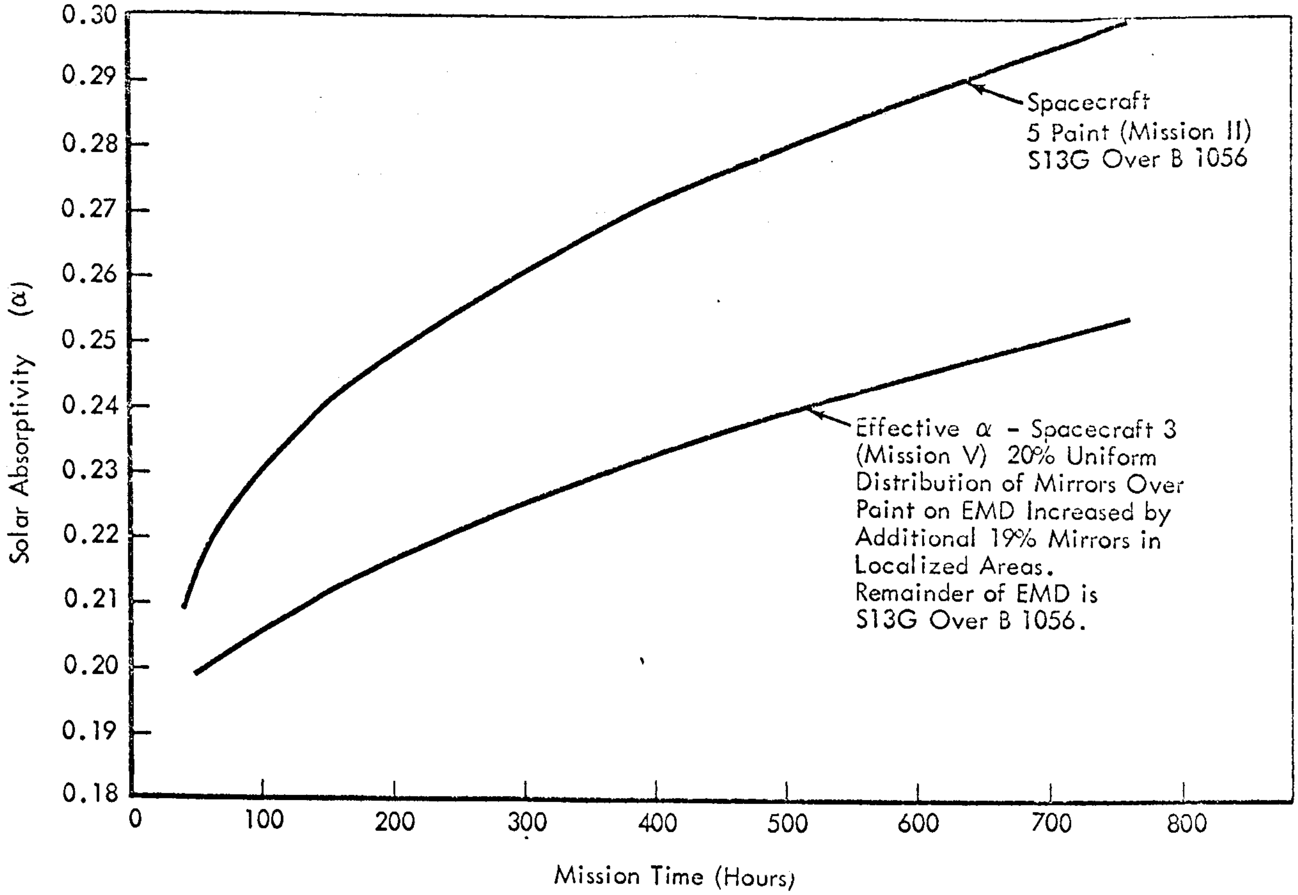


Figure 3-27: Comparison of Spacecraft 3 and 5 Solar Absorptivity

The radiation stability of these paints is measured in terms of the solar absorptance and infrared emissivity coefficients. The ratio of solar absorptance to infrared emissivity coefficients for each paint sample is obtained through the relationship

$$\frac{\alpha_S}{\epsilon_{IR}} = \frac{\sigma\,(T_{\text{PAINT}})^4\ \text{CORRECTED}}{S}$$

where: S = solar constant

σ = Stephan-Boltzman constant

(T paint corrected) = measured paint sample temperature by S/C telemetry and corrected for extraneous energy sources

α_S = solar absorptance coefficient

ϵ_{IR} = infrared emissivity coefficient

The ratio of solar absorptance coefficient to the IR emissivity coefficient is shown in Figure 3-28 for the thermal coatings included in this experiment. The data represented includes only the time duration covered during the primary mission; extended mission data will be reported elsewhere.

3.6.5 Thermal Design Difference Between Missions IV and V

Mission V presented a similar thermal situation to Mission IV, except the lower orbit introduced greater heating from emitted and reflected thermal radiation from the lunar surface. As with Mission IV, the high-inclination orbit precluded the spacecraft entering the Moon's shadow. The absence of a cooling period while the spacecraft is in the dark would raise the spacecraft temperatures to unacceptable levels. Also, mission requirements dictated orientation of the spacecraft on-Sun, or nearly so, between photo sites in the same orbit to conserve nitrogen. In addition, the photographic portion of the mission was longer than Missions I to III, inclusive;

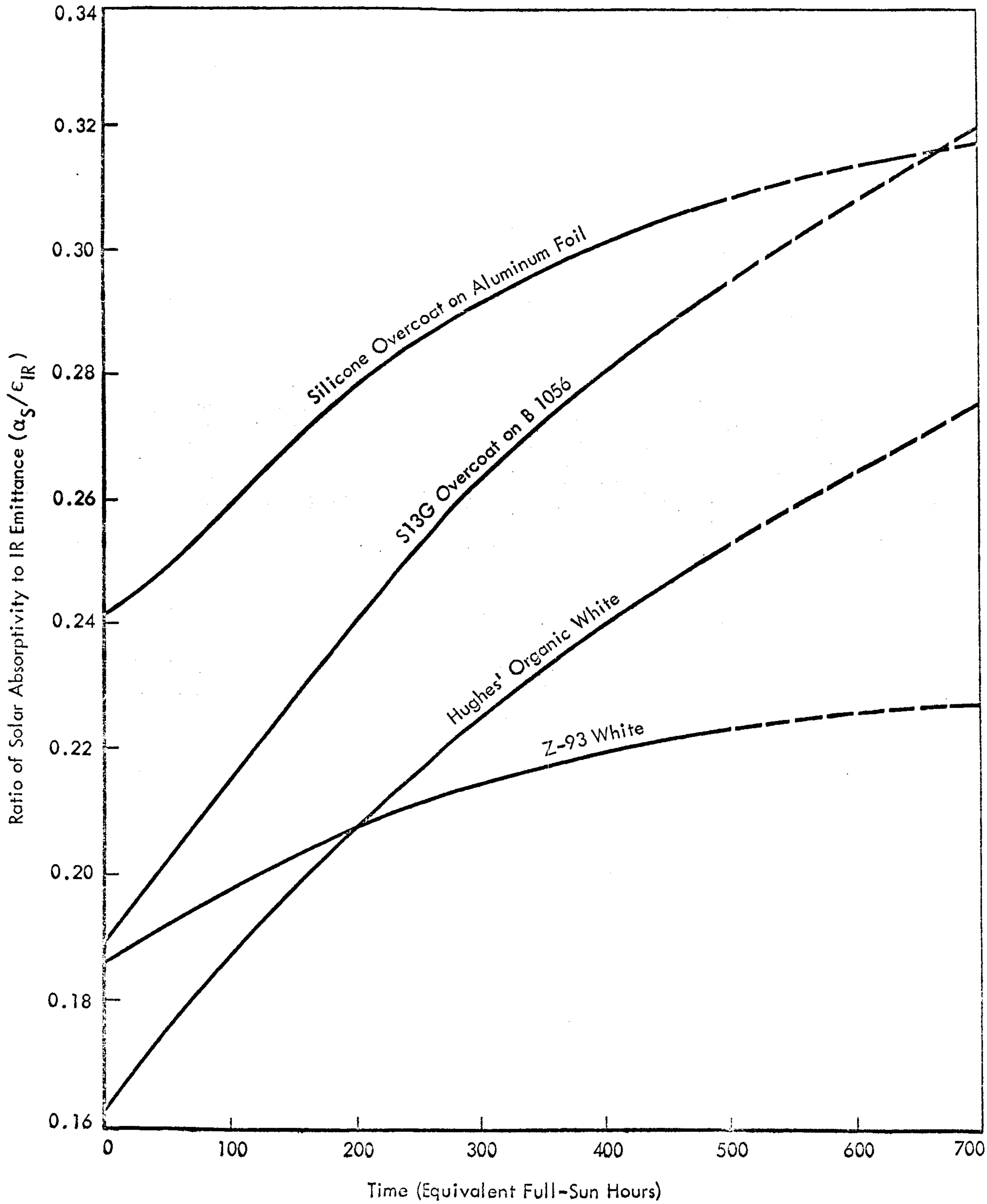


Figure 3-28: Thermal Coating Experiment: Ratio of Solar Absorptivity to IR Emittance Coefficients

hence, the paint degradation would cause unacceptably high photo subsystem temperatures and Bimat degradation. A thermal control paint that would have sufficiently low solar absorptance or a sufficiently low degradation rate to allow the mission to be flown as planned could not be identified. The very low solar absorptance of optical solar reflectors (OSR) compared to paint ($\alpha \approx 0.06$ versus 0.20 to 0.30) made it apparent that substantial reductions in spacecraft temperatures could be achieved through their use.

In Spacecraft, used in Mission IV, the OSR were arranged in a uniform pattern of 20% OSR and 80% paint, which was equivalent to using a paint with a lower solar absorptance. The lower orbital altitude for Mission V, the addition of a booster regulator, and an increase in battery charge current from 1.05 to 1.35 amps, required additional cooling. Based on Mission IV experience, this was achieved by increasing the concentration of OSR from 20 to 30% in the vicinity of the photo subsystem and the TWTA.

3.6.6 Thermal Performance

A history of Spacecraft 3 thermal activities is included in Table 3-6 for convenient reference.

Temperature History — The flight temperature history of Spacecraft 3 was generally similar to that experienced in previous missions. The temperatures for selected telemetry channels for the early cislunar phase of the mission are shown in Table 3-7. Typical temperatures encountered during other phases of the flight are shown in Figures 3-29 through -32.

3.6.7 Thermal Paint Experiment

Lunar Orbiters IV and V Thermal Paint Experiments -- Thermal coating experiments involving seven different paints were included on-board the Mission IV and V Lunar Orbiter spacecrafts. Designation of these coatings, along with a general description of the coating, is given in Table 3-8.

Use of a thermal paint in deep space was a fundamental aspect of spacecraft thermal control, providing the necessary radiation characteristics to maintain adequate temperatures, both on the painted surfaces and within the spacecraft.

The purpose of these experiments was to evaluate the stability of thermal coating radiation characteristics in a deep-space environment for long-time durations. The radiation characteristics of the painted surfaces are measured in terms of a solar radiation absorptance coefficient (α_S) and an infrared emissivity coefficient (ϵ_{IR}). The generalized relationship between temperature and the solar absorptance and infrared emissivity coefficients for an insulated flat plate with its normal directed along the solar vector is given by

$$T = \left(\frac{S}{\sigma} \cdot \frac{\alpha_S}{\epsilon_{IR}} \right)^{1/4}$$

where: S = solar constant

σ = Stephan-Boltzman constant

α_S = solar absorptance coefficient

ϵ_{IR} = infrared emissivity coefficient

T = equilibrium temperature of an adiabatic flat plate viewing the Sun.

Hence, unstable thermal coating radiation characteristics caused by the radiation environment in deep space will cause unstable temperature conditions on the Sun-exposed surfaces.

Comparison of Flight Test Data and Laboratory Measurements -- Initial radiation characteristics of each thermal coating were correlated between laboratory-measured data and flight test data. This correlation of data is illustrated in Figure 3-33. Hence, it can be seen that initial laboratory and flight test data compare favorably, considering the many variables encountered in the laboratory and flight data measurements and interpretation. The maximum deviation in α_S/ϵ_{IR} was approximately $\pm$ 0.02 for two of the paint samples, while the remaining paint samples had a mean deviation of approximately $\pm$ 0.01 in the initial ratio of α_S/ϵ_{IR}. This correlation is a means by which the absolute value of α_S/ϵ_{IR} for each paint can be interpreted in terms of confidence level.

Table 3-6:
History of S/C 3 Thermal Activities

(day: hr: min)	Thermal Activity	Remarks
213:22:33	Pitch to Sun	S/C Liftoff from Cape Kennedy
214:08:04	P/S heaters on	Maintain P/S temperature during cislunar cruise
214:14:34	TWTA on	
214:23:13	P/S heaters off	
215:04:10	P/S heater inhibit off	
215:05:21	TWTA off	
215:06:00	Engine burn	Midcourse correction
217:16:37	Pitch 96 degrees	Orbit injection
217:17:46	TWTA on	
217:17:50	Pitch to +40 degrees	
217:18:01	Solar eclipse off	(Day heaters on)
218:07:55	Acquire Sun	
218:10:30	TWTA off	
218:11:14	Pitch -176.9 degrees	Site A-1 photo
219:06:40	Solar eclipse on	
219:08:32	Pitch +71 degrees	First orbital transfer
219:08:46	Pitch to Sun	
219:13:43	Pitch to +171 degrees	Site A-6 photo
219:13:58	Pitch to Sun	
219:15:47	TWTA on	
219:23:14	TWTA off	
221:04:50		Second orbital transfer
221:05:10	Pitch to Sun	

Table 3-6 (Continued)

(day:hr:min)	Thermal Activity	Remarks
221:09:44	TWTA on	
221:09:50	Pitch +38 degrees	
221:15:51		Site A-14 photo
221:16:03	Pitch +38 degrees	S/C pitched +38 degrees between photo maneuvers for cooling
229:07:38	Pitch +41 degrees	S/C pitched to +41 degrees between photo maneuvers for additional cooling due to paint degradation.
231:02:43	Pitch to 30 degrees	To ensure sufficient electrical power for Bimat cut
231:04:20	Pitch to +38 degrees	38 degrees pitch angle for readout
239:06:45	TWTA off	
239:07:44	Pitch to +52 degrees	Cool S/C for period on Sun during gyro drift test
239:16:40	Pitch to Sun for gyro drift test	
239:21:12	Pitch to +52 degrees	Extended-mission pitch attitude

Table 3-7: Thermal Status Report

	Temperatures °F											
	L + 0	L + 2	L + 4	L + 6	L + 8	L + 10	L + 12	L + 14	L + 16	L + 18	L + 20	L + 22
GMT	213: 22:33: 00	214: 00:33	02:33	04:33	06.33	08:33	10:33	12:33	14:33	16:33	18:33	20:33
ST01 EMD/TWTA	54.4	42.9	43.3	43.6	45.1	46.0	47.3	47.6	47.6	70.7	73.1	74.0
ST02 EMD/batteries	56.7	56.3	55.0	61.7	66.7	69.0	69.9	70.8	70.8	72.2	73.2	74.1
ST03 EMD/IRU	56.8	62.7	64.0	64.9	66.8	67.7	68.6	69.9	70.4	70.9	71.3	71.8
ST04 tank deck	55.9	49.6	48.2	47.8	48.2	48.7	48.7	49.1	49.1	50.0	50.9	50.9
CT02 transponder	61.7	63.4	63.9	67.3	72.0	74.5	75.4	76.2	76.6	77.1	78.3	79.6
ET02 Battery 1	68.0	65.3	64.1	79.8	88.5	91.9	93.3	94.0	94.3	94.7	95.4	96.4
PT07 lower environment	56.3	52.0	50.0	48.8	48.1	46.6	51.0	53.4	54.4	56.1	56.6	57.8
PT08 upper environment	57.1	54.1	52.2	51.2	50.5	51.0	53.6	56.1	57.1	59.2	60.0	61.4
CT01 TWTA										152.5	156.3	157.3
Orientation from Sun	Launch	On Sun	On Sun	On Sun	On Sun	On Sun	On Sun	On Sun	On Sun	On Sun	On Sun	On Sun

COMMENTS. Photo system heaters on at Day 214, 08:04 GMT
TWTA on at Day 214, 14:34 GMT.

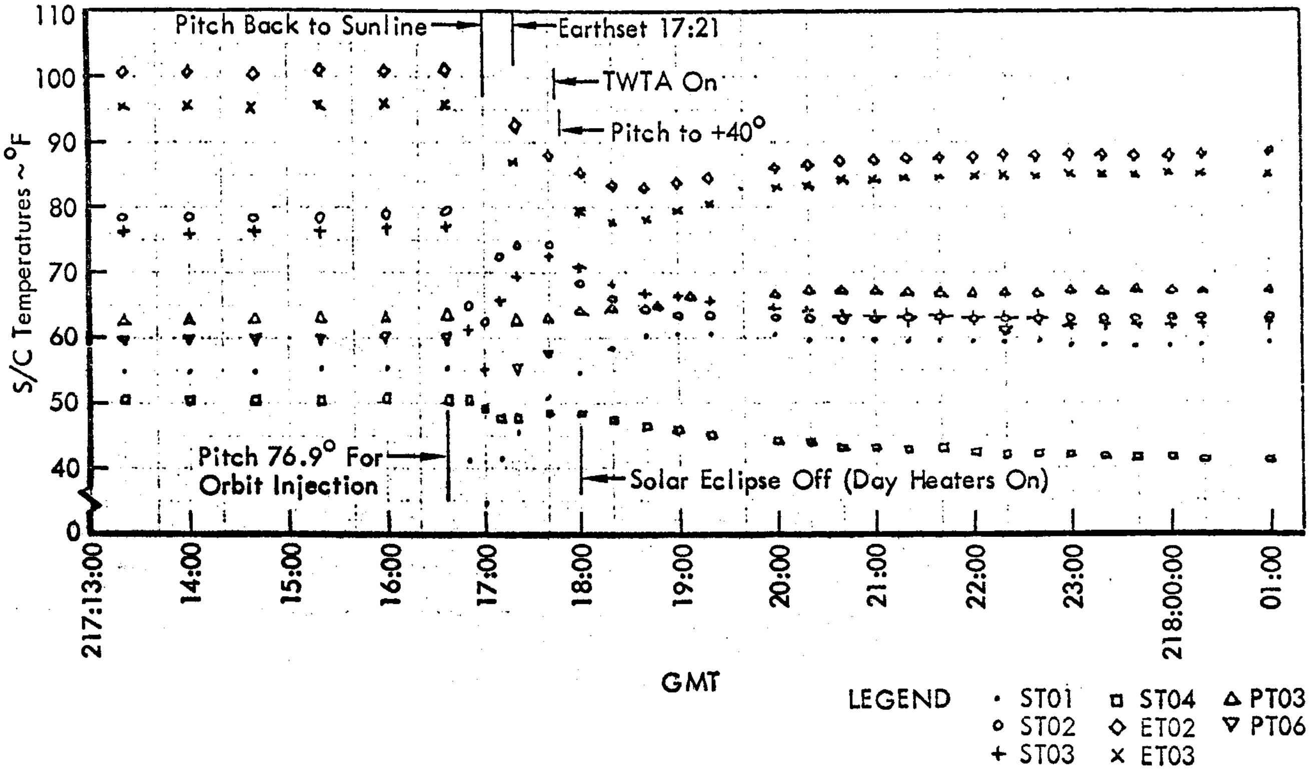


Figure 3-29: Typical Spacecraft Temperatures — Cislunar and Orbit Injection

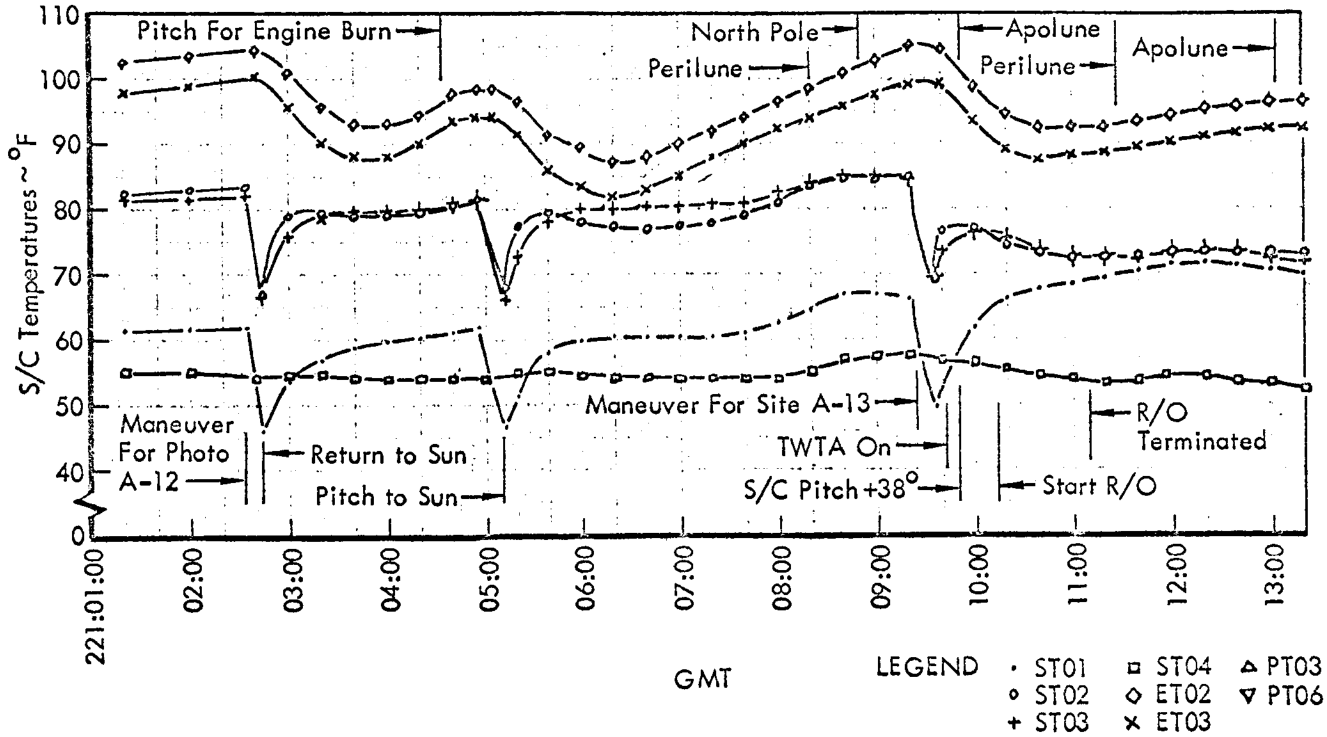


Figure 3-30: Typical Spacecraft Temperatures — Photo Maneuver, Engine Burn

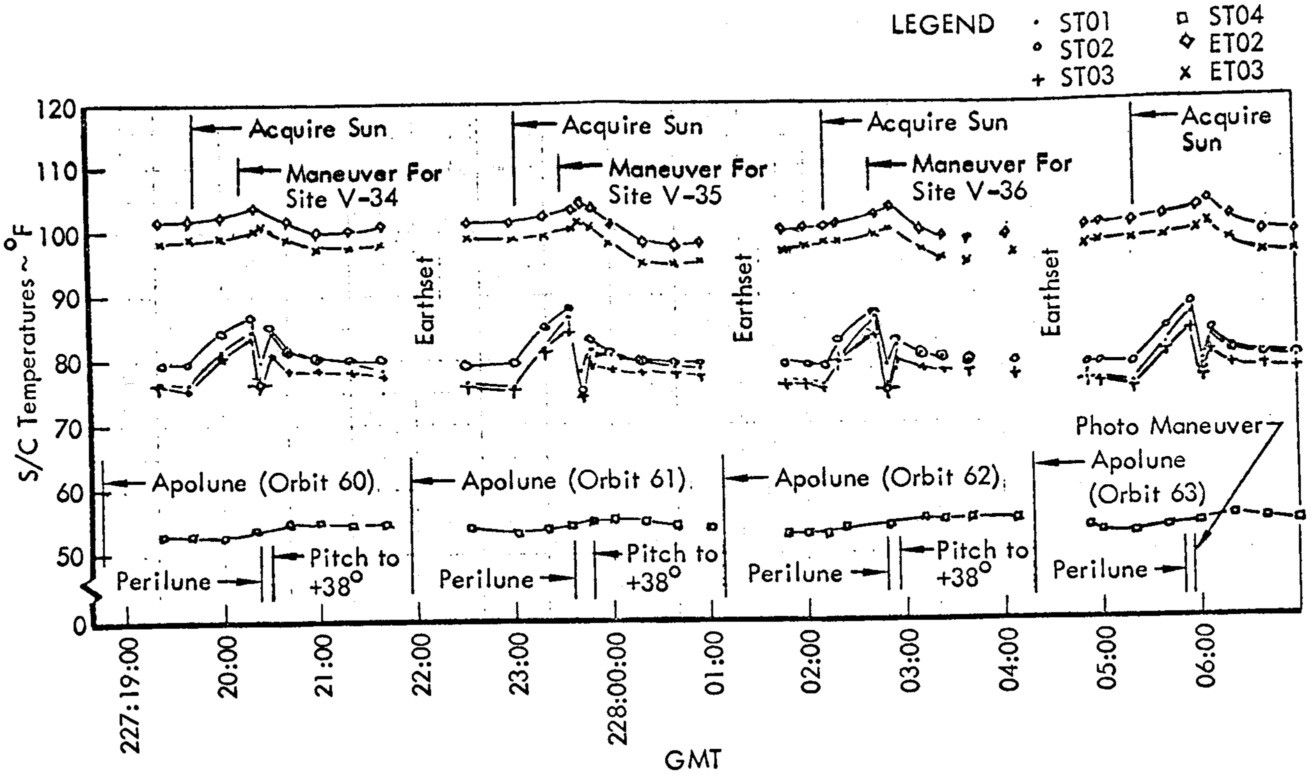


Figure 3-31: Typical Spacecraft Temperatures — Photo Maneuvers

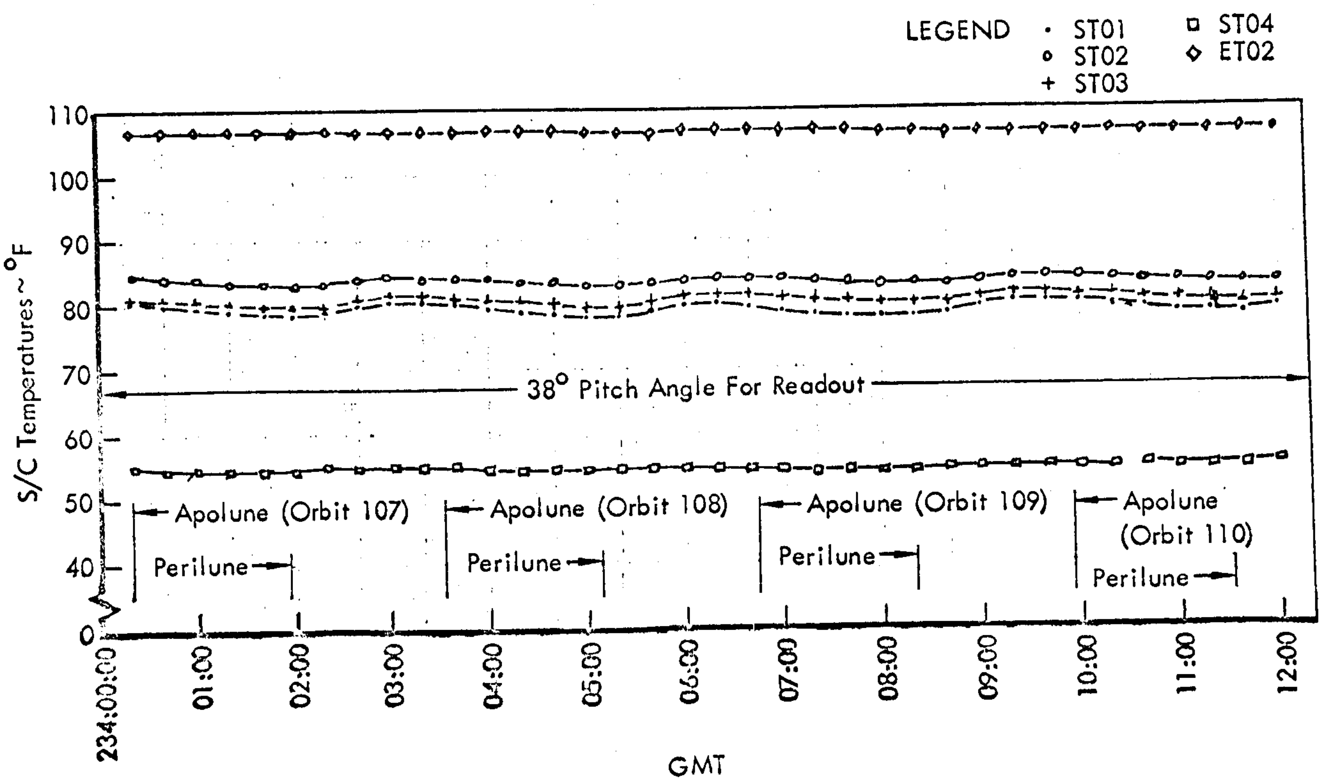


Figure 3-32: Typical Spacecraft Temperatures — Readout

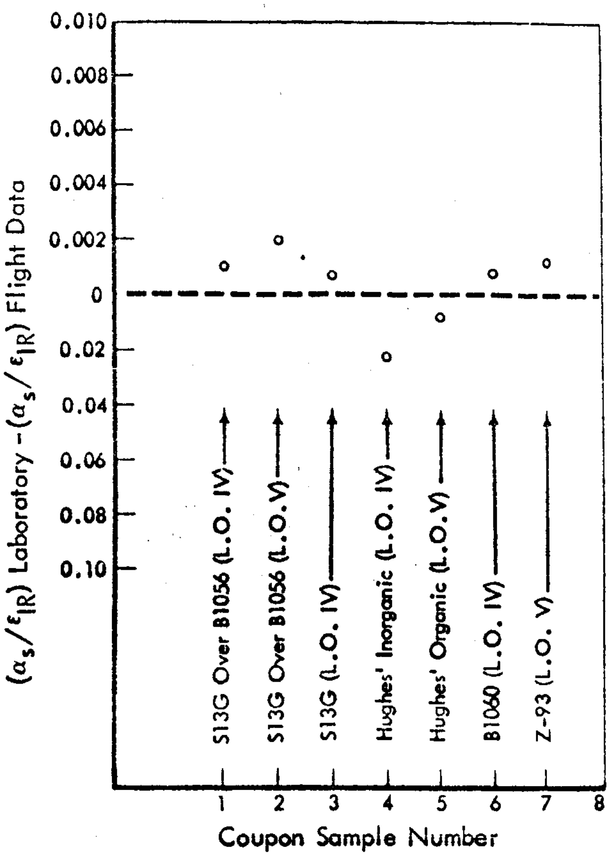


Figure 3-33: Comparison of Laboratory and Flight Data for Initial Thermal Coating Coupon (α_S/ϵ_{IR}) Ratio

Flight Test Data -- The experimental flight test apparatus has previously been described in the Lunar Orbiter IV mission report. Briefly, this apparatus consists of surfaces coated with the thermal paints of interest; each coated surface is instrumented with a thermistor that will allow measurement of the surface temperature.

The measured surface temperature data for each thermal coating must be corrected for extraneous energy sources, such as incident radiant energy from the nearby viewed spacecraft solar arrays. A more complete description of these correction factors is described in the Lunar Orbiter IV mission report. Both measured and corrected thermal coating temperatures are plotted in Figures 3-34 and 3-35 against equivalent full-Sun exposure time. The corrected temperature data given in Figure 3-35 include both extraneous energy source corrections and seasonal solar flux intensity corrections.

The temperature rise of each thermal coating coupon due to degradation in the thermal coating radiation characteristics is given in Figure 3-36. Several pertinent conclusions can be drawn from the data shown in this figure: (1) the various types of thermal coatings examined in these experiments exhibit a wide variation in degradation rates (e.g., the temperature rise of one thermal coating at 600 hours equivalent full-Sun exposure was 70°F, while another thermal coating at the same time period showed a temperature rise of only 24°F); (2) the inorganic-type thermal coatings appear to be more stable than their organic counterparts.

Measured Coating Radiation Properties in Flight -- The ratio of solar absorptance coefficient to infrared emissivity coefficient (α_S/ϵ_{IR}) for each of the thermal coatings included in the L.O. V experiment is given in Figure 3-37. The data indicate that Z-93 thermal coating was more stable than all of the other thermal coatings tested, including the L.O. IV experimental coatings.

The instantaneous degradation rate of each thermal coating is shown in Figure 3-38. The data indicate that the degradation rate of the organic coatings is still high at 700 equivalent full-Sun exposure hours, while the inorganic type coatings had a very low degradation rate at the same time period. Initially, the ratio of degradation rates for the organic coatings compared to the inorganic coatings was approximately 2.1, while this ratio had increased to approximately 7.0 after 700 equivalent full-Sun exposure hours.

Table 3-8: Coating Descriptions		
Coating Designation	S/C Test Bed	General Paint Description
S-13G Overcoat on B1056	L.O. IV & V	S13G is described below. B1056 is ZnO (New Jersey Zn SP500) in a Silicone resin (RTV-602).
S-13G	L.O. IV	ZnO (modified with potassium silicate) pigment in a silicone (RTV-602) resin.
B1060	L.O. IV	ZnO (modified with potassium silicate) in a silicone resin (RTV-602). Boeing proprietary catalyst incorporated.
Hughes' Inorganic White	L.O. IV	TiO_2 pigment in a potassium silicate binder (Sylvania PS-7).
Hughes' Organic White	L.O. V	Aluminum silicate pigment in a silicone binder (RTV-602).
Silicone over aluminum foil	L.O. V	Silicone (RTV-602) coating on pure aluminum foil.
Z-93	L.O. V	ZnO pigment incorporated in a potassium silicate binder (Sylvania PS-7).

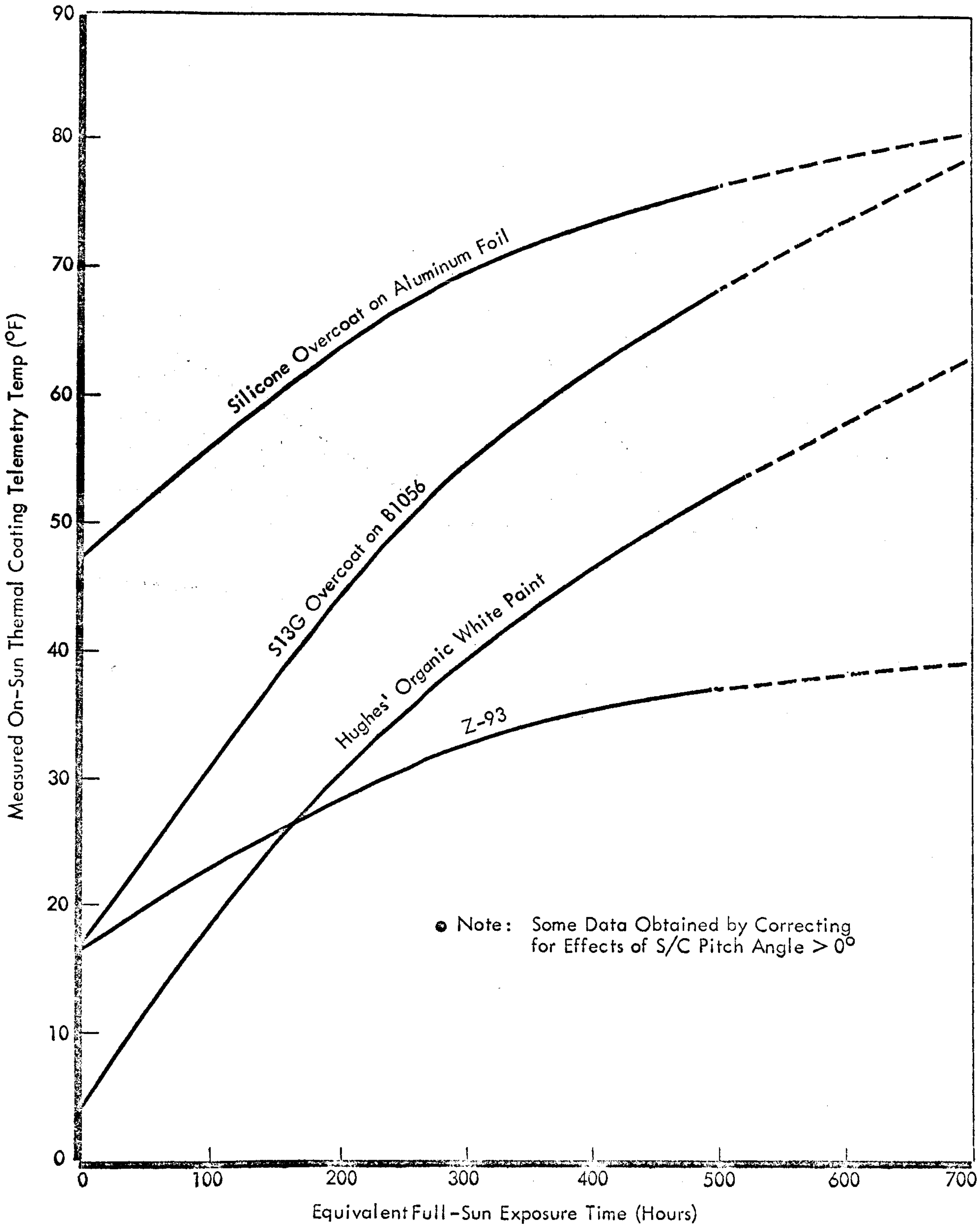


Figure 3-34: Measured on-Sun Thermal Coating Coupon Temperatures

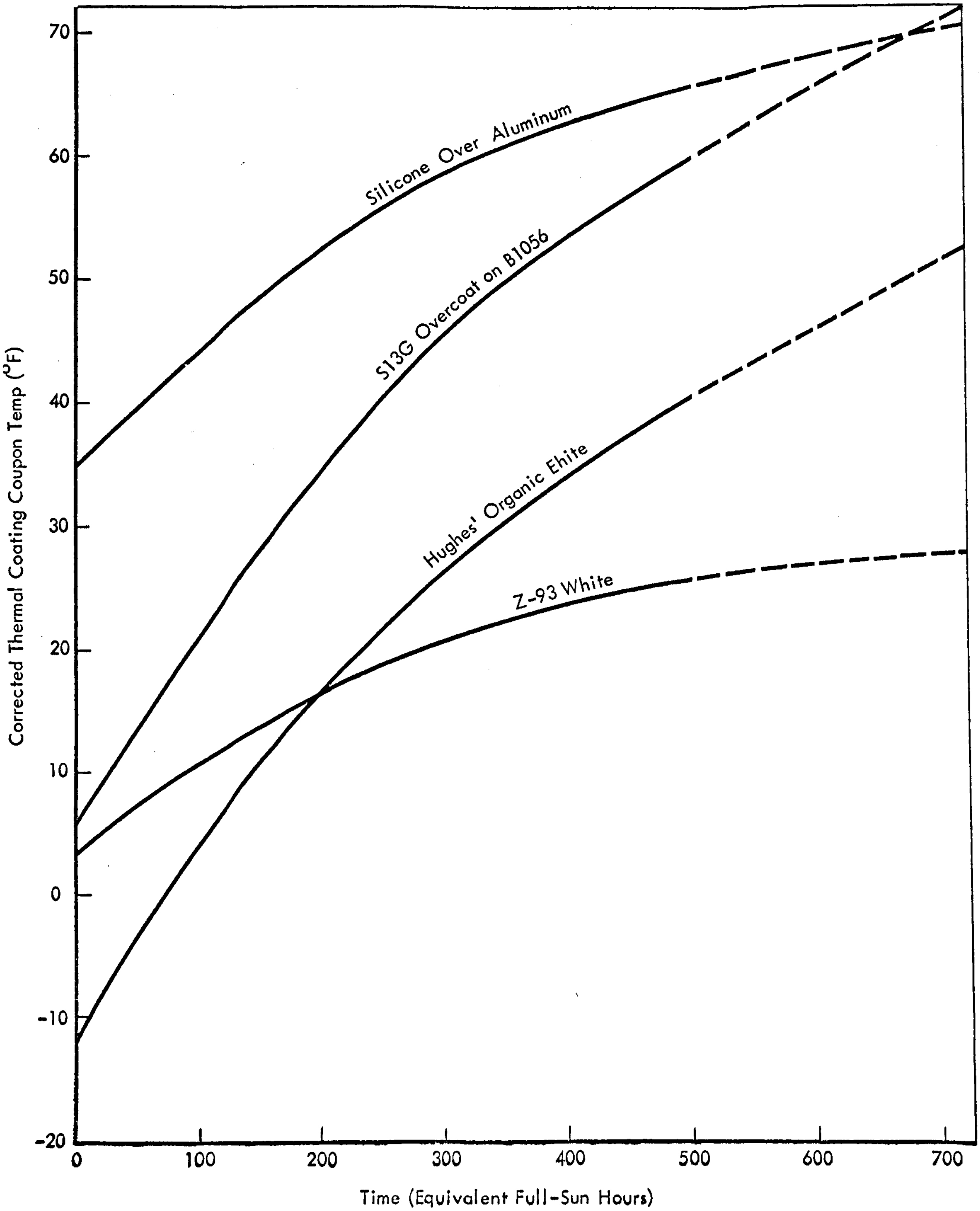


Figure 3-35: Corrected Thermal Coating Coupon Experiment Temperatures

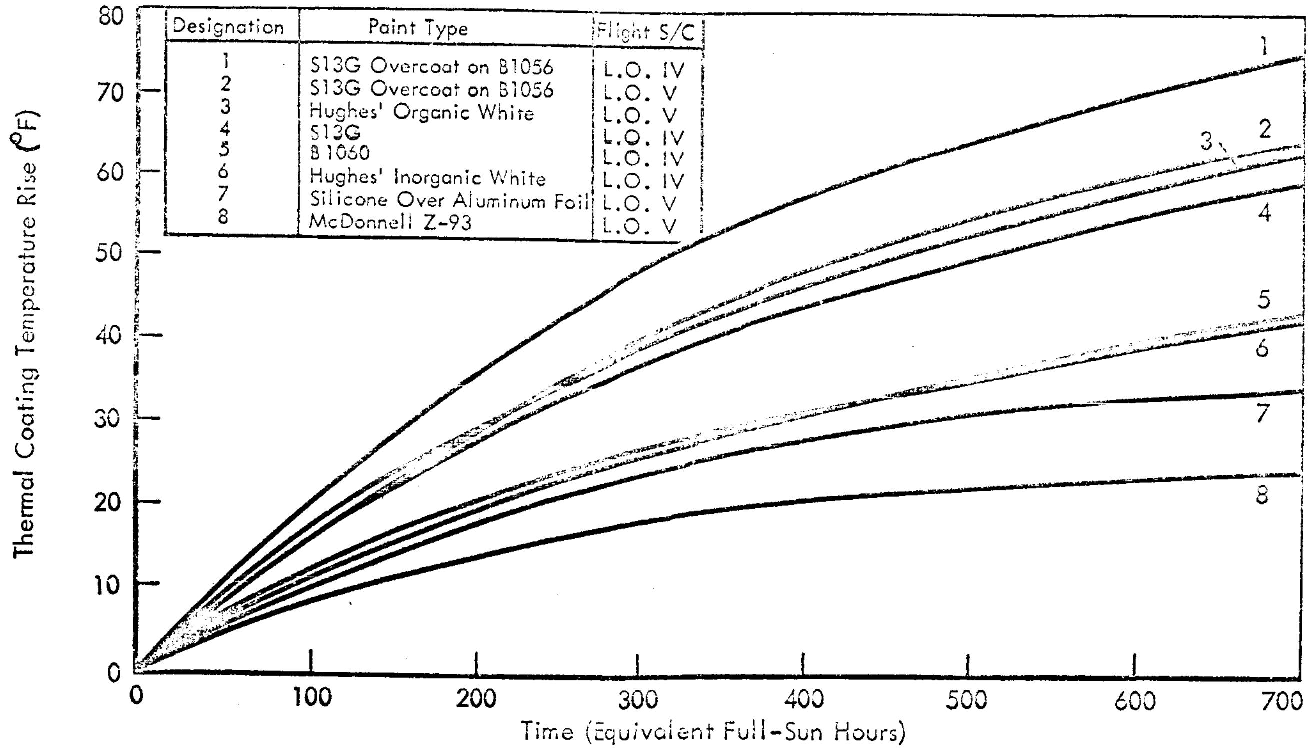


Figure 3-36: Effect of Thermal Coating Degradation on Sample Temp Increase from Mission Start

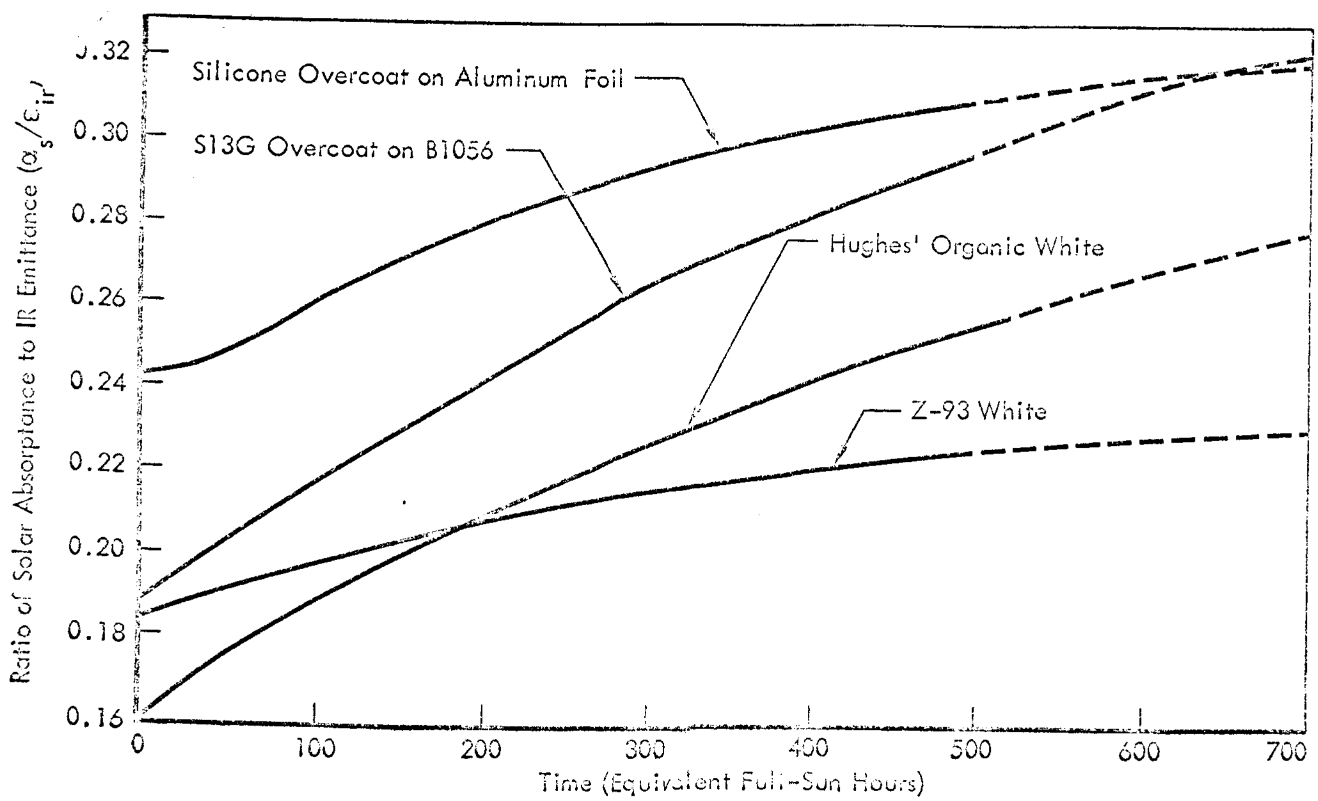


Figure 3-37: Thermal Coating Experiment — Ratio of Solar Absorptance to IR Emittance Coefficients

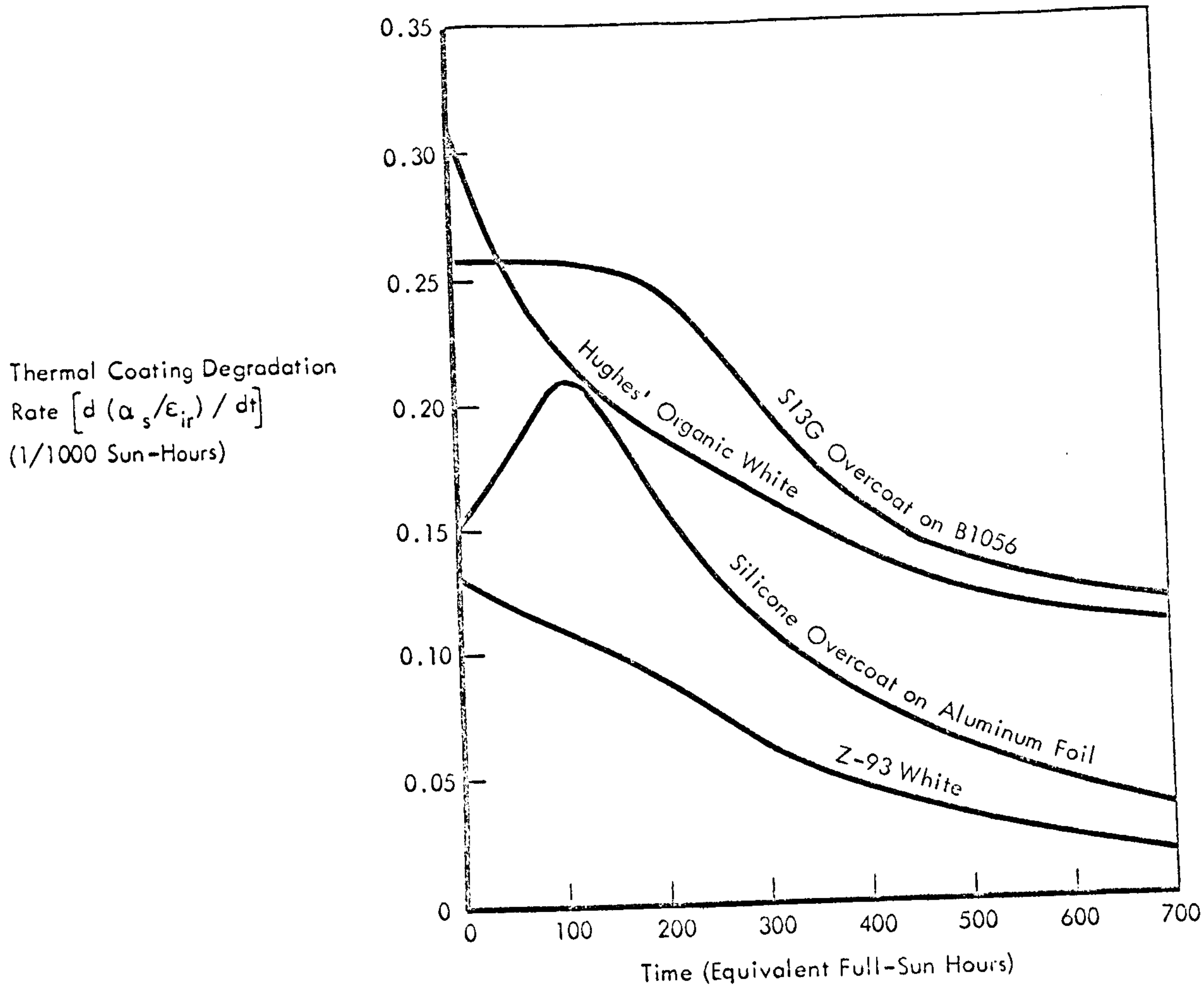


Figure 3-38: Experimental Thermal Coating Degradation Rates

4.0 Ground Data System Performance

The Lunar Orbiter ground data system provides facilities and equipment required to receive, record, process, and transmit data and commands between the Space Flight Operations Facility (SFOF) and the spacecraft. In addition, all facilities necessary to sustain mission operations were provided by means of a complex, consisting of three primary Deep Space Stations (DSS), the SFOF, and the ground communications system. Separate facilities were provided at Eastman Kodak, Rochester, N.Y., and at Langley Research Center, Hampton, Virginia, to process and evaluate the photo data obtained.

4.1 SPACE FLIGHT OPERATIONS FACILITY (SFOF)

The SFOF provided the mission control center and facilities to process and display data to support operational mission control. Facilities were provided for the ground reconstruction equipment and for analysis of the reconstructed lunar photographs; there were also facilities for reproduction and distribution of operational data and for microfilming all computer program output. Overall performance of the entire SFOF data system was very good.

4.1.1 Computer and Communications Complex

The telemetry processing station (TPS) and the internal communications system at the SFOF provided tracking and telemetry data from teletype and the high-speed data line to the SFOF computers and teletype data to the operations areas. The computer complex provided telemetry data processing, tracking data processing, command generation, command verification, and compilation and transmission of the mission sequence script and station predicts. The central computing complex consists of four computer strings each containing an IBM 7094 computer coupled with an IBM 7044 input-output (I/O) processor through an IBM 1301 disk file memory and a direct data connection (DDC). The fourth string is not fully operational in a real-time sense, but was used for off-line functions involving orbit determinations where other computers were not available.

All four operating computer strings, listed below, were used to support Mission V.

Computer Strings	Total Hours	Dual Mode 2
X	355	350
Y	366	306
W	261	94
V	18	18

The total amount of Mode 2 time used was 1,000 hours. Dual Mode 2 was used during all critical phases and throughout most of the mission due to the tight mission constraints. Dual Mode 2 computer strings were not always up as a back up for each other.

4.1.2 System Software

The software system for Mission V contained changes from the Mission IV software. The system was successfully demonstrated prior to the Mission V training exercises, and performed exceptionally well with no serious software failures.

The SFOF mission independent software performed satisfactorily throughout Mission V, with no exceptions. The chronic occurrence of internal restarts on the 7044's caused by Communication Error 01 which plagued Mission IV was corrected with no significant recurrence. (The Mission IV problems are described in NASA Document CR 66498, *Lunar Orbiter IV Final Report - Mission System Performance)*

4.1.2.1 SPAC Software System Performance

The SPAC software consists of the IBM 7094 computer programs utilized to monitor the telemetry from and to predict the status of the spacecraft subsystems. It also consists of a program that prepares and simulates command sequences to be transmitted to the spacecraft computer and of a program that coordinates mission planning. Changes to the SPAC programs for Mission V were limited to the thermal program, which was expanded to allow greater capabilities in status reporting and plotting. There were no failures in any of the SPAC computer programs.

Table 4-1, a tabulation of all SPAC computer program executions, divides unsuccessful executions into two groups. Input errors include mispunched input cards, incorrect messages, and option switches entered from the input console. System errors consist of system hardware and software failures. SPAC software errors include accessing a bad master data table, a bad common environment on disk, and all SPAC software. Hardware errors include card reader problems, sense line errors, and communication errors while reading cards.

4.1.2.2 FPAC Software System Performance

Between Missions IV and V, a number of modifications were incorporated in the FPAC software system. Some of these changes were dictated by the necessity to correct errors discovered through the use of the software and its changes in the previous flight; others were made to improve the efficiency and speed of computation, provide additional computational tools to the analysts, and eliminate hand copying data for command conference forms. The performance of all FPAC computer programs during Mission V was satisfactory. A late change to user programs EVAL, involving the addition of a new link, was available on only one of the two computer strings. This string was largely at the disposal of orbit determination analysts; hence, it was not used as extensively as it might have otherwise been. A brief description of the changes that were made follows.

Flight Path Control Programs -- User programs [illegible] and INTL were modified to allow more efficient and rapid computation of occultation and range data. Changes included were that greatly simplified the required inputs and made the program operation more automatic. Chances of error were reduced so that the first run was usually acceptable.

A much-needed coordinate transformation package was added to subprogram PIG, which appears in user programs INML and MDLL. This option greatly aided the guidance analysts as it allows the inputting of a state vector in either conic or cartesian coordinates. An additional advantage brought about by this modification was a better interface with these programs and the orbit determination program (ODPL).

In the links COMD1 and XMAN1, the pseudo Sun and pseudo Canopus vectors were redefined to correct a subtle error discovered during Mission IV. It was necessary to replace the IRU roll and pitch axes with the sun sensor and the Canopus tracker axes for the computation of these vectors. The effects of this computational error could be cancelled by a work-around procedure, but the chance of analyst error was great.

The data storage region in GENL was changed to update several of the physical constants not normally input to the flight path control programs. Problems were encountered during Mission IV when the outdated values nominally in GENL were mistakenly used in computations.

A change in the printout of the pre- and post-injection state vectors and of the thrust directions cosines in of-dates coordinates was required. The correction made it possible to properly input the thrusting maneuver to PRDL (to generate predicts) and to TRJL (to check the injection maneuver).

User programs GCPL and EVAL were modified to correct the output of longitude and latitude points describing the photo frame edges. The previous form left a large region on each edge not calculated, making it difficult to plot photo coverage with any degree of accuracy.

A new link, FOTO9, was added to user program EVAL. This link has the ability to process the

Table 4-1: SPAC Program Execution

Prog	Total Runs	No. of Successes	System Errors: H/W Failures	System Errors: S/W Failures	Input Failures
CEUL	486	486	0	0	0
DATL	427	409	1	8	9
TIML	294	282	0	0	12
GASL	90	88	2	0	0
COGL	245	243	0	2	0
SEAL	57	55	2	0	0
TRBL	113	110	1	0	2
COOL	10	8	0	0	2
UTAB	20	20	0	0	0
QUAL	203	203	0	0	0
SGNL	15	15	0	0	0
SIDL	16	16	0	0	0
HUBL	29	29	0	0	0
CORL	4	4	0	0	0
TOTAL	2,009	1,968	6	10	25

Percentage Calculations

	Total Runs	No. of Successes	H/W Failures	S/W Failures	Input Failures
TOTAL	2,009	1,968	6	10	25
%	100	97.95	0.3	0.5	1.25

photo data necessary for a command conference, and then print out the data in a form acceptable for distribution, thus eliminating the need to hand copy the data.

Tracking Data Editing Programs (TDPX and ODGX -- Two changes were made to the TDPX source deck for Mission V:

- DSS-51 (Johannesburg) was defined as an s-band station;
- Tracking Data Formats 01, 02, 03, 05, 06, 07, 09, and 13 were made available, (Formats 05, 06, 07, 09 were made to include the doppler resolver data field).

Two changes were made to the ODGX source deck:

- Ranging data processing was made possible when angle data was flagged bad (DC2 = 1);
- All ranging data points with a good data condition code (DC6=0,2) are processed and placed on the ODP file.

The above changes worked as expected.

Two ODGX problems were encountered during the mission. The first problem was the discovery early in cislunar phase that an error existed in the ODGX computations concerning the doppler resolver data. A minus sign had been programmed instead of a plus sign, resulting in an apparent increase in the doppler data noise level instead of the expected decrease. A source deck patch was prepared and used successfully throughout the remainder of the mission. The second problem was a recurrence of a difficulty noticed in prior missions (i.e., the sporadic inability of ODGX to read the TDPX master file properly). The random occurrence of this problem has so far prevented a solution.

Orbit Determination Program (ODPL) -- Two different Benchmark software systems were used for orbit determination during Mission V. The "standard" Mission V software system, F.1, was used exclusively during cislunar phase; the "non-standard" Benchmark F.3 software system was used for 95% of the orbit determination computations during lunar orbit phase. The changes made to ODPL incorporated in the Benchmark F.1 system were:

- "Obtain plots" feature -- Program user may obtain residual plots immediately after any iteration rather than doing another iteration;
- Improved JPL-Boeing mode communications -- This corrected a program logical error discovered during Mission IV;
- Change DSS-51 from L-S band to S-band -- Required because of a DSN change to station equipment;
- Input higher order harmonics -- Lunar harmonic coefficient input was expanded from Order and Degree 4 to Order and Degree 10 in response to LRC provided lunar models;
- Ranging unit plot scale -- Corrected an annoying program error concerning labeling of the RU plot scale;
- Correct DSS-62 printout -- Due to logical error, all DSS-62 tracking data was labelled 13, later changed to print out DSS-62;
- Change LA9 error routine -- A logical programming error was corrected that allowed the program to continue with wrong input data after three successive input errors.

The Benchmark F.3 system included all the above ODPL changes plus the following additional changes:

- Report generation -- a new link was added to automatically printout and format the orbit determination summary report and Goddard state vector report, which was handwritten prior to this change;
- An additional switch option was added to a complete termination of the program in Boeing input link B9;
- Updated time polynomial coefficients -- a new set of coefficients to convert from WWV time base to universal and ephemeris times was included in the source deck.

Changes to both Benchmark systems worked as planned.

One significant program problem reoccurred during this mission. A logical program error in the SPACEL link of ODPL causes the program to abort when doing a mapping if the mapping time happens to be a particular value. Prediction of these troublesome mapping times is possible but not feasible, and this problem generated a crisis during the mission. For future applications, this problem should be eliminated.

4.1.3 Ground Reconstruction Equipment

GRE Serial Number 02 was operated at the SFOF. The equipment occupied the same area used in Mission IV. Both priority readout and final readout were supported. The GRE at the SFOF provided mission control data in the following areas:

- Photo quality evaluation;
- Exposure evaluation;
- Photo subsystem performance analysis (e.g., Bimat defects and video dropouts);
- Targeting and coverage analysis.

This unit was an invaluable supplement to the station GRE's, because some of the photography could be evaluated in real time by operations and mission control personnel. This direct analysis provided a running calibration of the station reports. The SFOF unit was most useful when assembled pictures were required, as in Item 4, or when the photo quality of a small feature was in question.

4.1.4 Ground Communications System

The ground communications system provides for the transmission of voice, teletype, and high-speed data between DSIF sites and the SFOF. One high-speed data line (HSDL), one voice line, and three teletype lines are provided between each station and the SFOF. The primary source of spacecraft performance telemetry data is the HSDL. One or two teletype lines are provided as a backup for the HSDL depending on the priority assigned to the second teletype line. The remaining TTY lines are used for tracking data, command transmission and verification, and administrative data.

Ground communications were exceptionally good during Mission V. There were fewer instances of circuit failures than in previous missions, no failures during critical periods, and only two periods in which all TTY and HSDL communication were lost. These occurred at DSS-62 for periods of 5 and 10 minutes. During the former period the DSS-62 voice circuit was also down. The transmission of one programmer map required a voice verification due to a line failure. During all other communications failures there was adequate backup capability.

High-speed data line failures averaged about ten minutes per failure from all stations. Actual outages ranged from 3 to 32 minutes excluding a 210-minute failure at DSS-62 (this latter case was not included in the average figure). TTY outages also averaged about 10 minutes; times ranged from 1 to 60 minutes. There were few voice line outages ranging from 1 to 13 minutes. The following table shows ground communication complex down time as a per cent for each station.

	HS	TTY	Voice
DSS-12	0.30%	0.01%	0.09%
DSS-41	0.47%	0.30%	0.20%
DSS-62	0.79%*	0.60%	0.36%

*Does not include one 210-minute failure. The corrected figure is 2.27%.

4.1.5 Deep Space Stations

The Deep Space Stations received tracking, spacecraft performance, and photo data from the spacecraft. All stations performed satisfactorily. Priority readout activity was higher than previous mission and required additional GRE film and shipping containers, which were supplied promptly to all stations.

4.1.5.1 DSS-12

All Lunar Orbiter MDE performed satisfactorily throughout the mission with the exception of two incidents involving the test transponder and GRE-04. The test transponder failed during the station countdown on August 9; however, two-way lock was maintained and the count continued. A replacement transponder was received from DSS-71 and installed prior to the next pass; no tracking time was lost.

The GRE-04 camera power supply fuse blew at the start of Readout 129. Thirteen framelets of film were lost on this GRE; however, there was no loss of mission coverage as DSS-41 was prime at the time.

There were three significant transmitter failures during the mission. Uplink was lost on August 10 when the transmitter heat exchanger door was slammed causing a relay to kick out. Although this incident occurred during a readout down link was maintained, and there was no data loss. On August 15 the transmitter failed when a filament wire and a high voltage wire shorted. DSS-41 was immediately brought up two-way with no data loss. On August 23 the exciter power supply failed. Down link was maintained but readout was delayed for 1.5 hours until Station 41 could view the spacecraft. By that time the problem was corrected and readout was continued.

4.1.5.2 DSS-41

Lunar Orbiter MDE performed satisfactorily throughout the mission. There were no failures or anomalies. The FR 900 recorder failed prior to the final two readouts on the last day of the mission.

All DSS equipment also performed satisfactorily with no significant failures. Equipment failures were limited to the normal maintenance type. No data were lost.

4.1.5.3 DSS-62

The MDE performed acceptably throughout the mission although there were several minor failures that were immediately corrected without significant data loss. There was an intermittent RF leakage in the test transponder cabling which caused the B•T error test to be out of tolerance. This problem disappeared by the end of the mission.

The station equipment was also satisfactory. There were a few incidents in which the transmitter failed; however, they were all corrected within a few minutes.

5.0 Lunar Environmental Data

5.1 RADIATION DATA

During Lunar Orbiter Mission V, the radiation dosimetry measurement system functioned normally and provided data on the Earth's trapped radiation belts and on the radiation environment encountered by the spacecraft in transit to and near the Moon. Dosimetry data are recorded in Table 5-1.

The cassette dosimeter, Channel DF04, indicated a total of 0.75 rad received passing through the outer Van Allen belts. This total contrasts with the 5.5 rads received by Orbiter IV on a similar trajectory through the outer belt. For the Orbiter V launch, however, magnetic conditions were much calmer than in May. Evidently, the outer electron belt was not enhanced during the August launch phase.

During the cislunar flight and throughout the photographic mission, solar activity remained at a low level and only the normal combination of galactic cosmic ray background and dosimeter dark current was recorded. A summary of dosimeter changes is given in Table 5-1.

5.2 MICROMETEOROID DATA

During the photographic portion of Mission V, one micrometeoroid hit was recorded, at Day 221, 04:57:28 GMT on Detector 6. Refer to Figures 5-1 and 5-2 for further details on spacecraft location at time of impact.

Table 5-1: Dosimeter Record

GMT	Detector	Reading
214:00:19:26	DF04	0.25
214:00:33:15	DF04	0.50
214:01:12:48	DF04	0.75
218:11:44:30	DF04	1.00
223:05:35:05	DF05	0.50
227:10:49:58	DF04	1.25
236:13:53:31	DF04	1.50
236:23:17:14	DF05	1.00

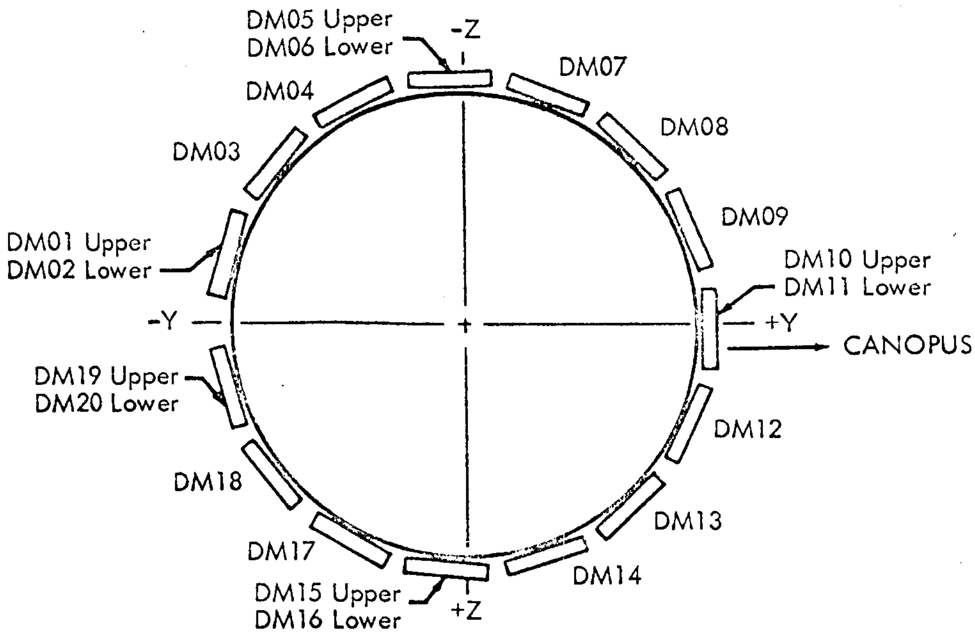


Figure 5-1: Micrometeoroid Detector Locations

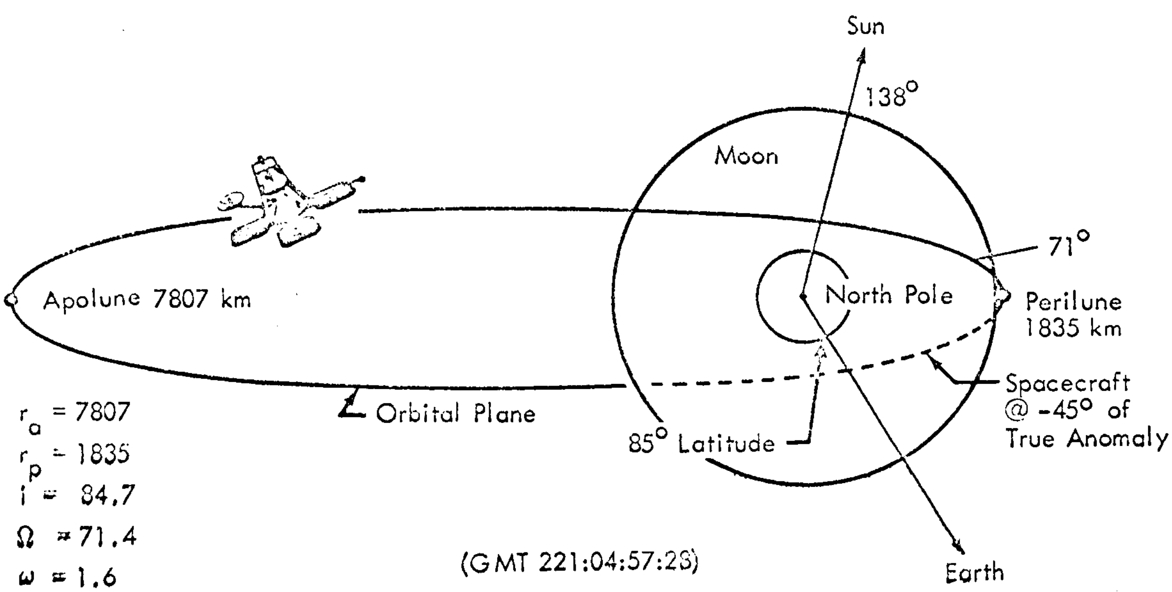


Figure 5-2: Location of Micrometeoroid Impact-Dectector 6

Summary of Lunar Orbiter V Anomalies

The following paragraphs discuss the principal malfunctions occurring during Mission V.

MOMENTARY LOSS OF VIDEO

During Readout Sequence 001, several momentary video dropouts were noted by DSS-12 and DSS-62. Attempts were made to isolate the problem by duplicating the phenomenon using Photo Subsystem PS-1A. It was determined that the problem could be repeated to some extent. As a result of the PS-1A test, it was decided to increase spacecraft photo subsystem temperatures. Temperatures were increased to the point where video dropouts were significantly reduced, and yet a proper moisture content was maintained to prevent Bimat dryout. There was no effect on the data as those areas of dropout during priority readout were recovered during final readout.

LOSS OF TWO-WAY LOCK WITH DSS-12

During command activity at DSS-12, two-way lock was lost with the spacecraft. Two-way lock was lost simultaneously with the completed execution of ranging off. It was speculated that the high-static phase error caused the loss of uplink, followed immediately by the loss of downlink. The spacecraft was reacquired after 13 telemetry frames. The anomaly did not repeat; consequently, no work-around was required nor was there a loss of mission data.

TERMINATION OF VIDEO DUE TO RF INTERFERENCE

On August 9, 1967, readout was terminated due to extremely noisy video. In addition, telemetry was noisy, with all spacecraft sub-systems showing bad data. It was discovered that the DSS had their test transponder turned on at the time. The test transponder was turned off and readout continued. No mission data were lost due to the anomaly.

PREMATURE BIMAT EXHAUSTION

"Bimat cut" command execution was scheduled to occur August 19 at approximately 03:24 GMT. "Bimat clear" was received 5 minutes prior to execution of the "Bimat cut" command. Investigations indicated that the Bimat roll used for this mission was 5 feet short of that normally used. Two wide-angle frames were not processed as a result of premature Bimat exhaustion.

FILM BREAKAGE DURING REWIND

On August 27, 1967, at 16:10 GMT, the 70-mm film separated between the processor/dryer and the readout gate. The first indication of film separation occurred when the readout looper contents jumped from 5.1 to 54.3 inches. In addition, the attitude control subsystem indicated the spacecraft experienced some significant disturbances at the time the readout looper went to mechanical full, indicating a mass movement took place in the spacecraft. Camera operation is a secondary consideration during extended mission; therefore, film breakage had no effect on either the extended mission or data.